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SIMULATOR STUDY OF PILOT-CONTROLLED LUNAR TAKE-OFF AND RENDEZVOUS

by Charles P. Llewellyn

Langley Research Center
Langley Station, Hampton, Va.

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SUMMARY

23254

A three-degree-of-freedom, fixed-base simulation study of pilot-controlled lunar trajectories from lift-off through rendezvous with a space station orbiting at a 100-nautical-mile altitude has been made. The results of this planar study have shown that a pilot can visually determine his launch time and effectively manually control both vehicle attitude and main-engine cut-off to arrive at the proper altitude and position to successfully and efficiently initiate and complete a rendezvous maneuver. It has also been shown through the use of three trajectories having coast angles of 24°, 90°, and 180° that a launch window of about 4 minutes is available. An early launch capability extended the launch window to about 5 minutes and alleviated some of the launch on-time problems.

Author

INTRODUCTION

Man will participate in some of the lunar missions, and the degree of participation will depend on his demonstrated ability to perform various tasks efficiently and with a high degree of reliability. Examination of the tasks which man might perform, the development of procedures, and the demonstration of these procedures must be performed well in advance of the actual mission. A large part of this work will be done on simulators. Some piloted studies of this type already have been made for the guidance and control of various phases of a lunar mission, including rendezvous, lunar landings, and aborts. (See refs. 1 to 4, for example.)

Another important area of investigation is that of launch from the lunar surface to a direct rendezvous with a lunar orbiting space station. This is a critical task due to the limitations placed on the relative positions of the satellite and ferry vehicle at launch, and the fuel available to perform the task. The purpose of this investigation was to determine if a pilot, by commanding vehicle attitude and main engine thrust level, could accomplish a successful and reasonably efficient lunar take-off and direct rendezvous with an orbiting space station in a 100-nautical-mile orbit about the moon with the use of relatively few instruments and certain visual cues. Also the aim of this study was to find a means whereby the launch window could be increased and the launch on-time problems alleviated.

SYMBOLS

F	control force, lb
g_e	gravity at surface of earth, 32.2 ft/sec^2
h	altitude above moon's surface, ft or int. n. mi.
I_Z	moment of inertia about vehicle pitch axis, slug-ft^2
I_{sp}	specific impulse of rocket, 424 $\frac{\text{lb-sec}}{\text{lb}}$
l	distance along longitudinal axis from reaction control jets to vehicle center of gravity, ft
m	vehicle instantaneous mass, slugs
$\dot{m}$	time rate of fuel consumption, slugs/sec
m_a	total fuel used by pilot, slugs
m_i	total fuel required for ideal nominal on-time launch and rendezvous, slugs
R	line-of-sight range from satellite to ferry vehicle, ft or int. n. mi.
$\dot{R}$	time rate of change of line-of-sight range, ft/sec
r	radial distance from center of moon, ft
$\dot{r}$	vehicle velocity component along radius vector, ft/sec
$r\dot{\phi}$	vehicle velocity component normal to radius vector, ft/sec
T	main engine rocket thrust along vehicle longitudinal axis, lb
t	time, sec
V	total vehicle velocity, ft/sec
W	weight of vehicle on earth, mg_e, lb
X,Y	vehicle longitudinal and vertical axes through vehicle center of gravity, respectively
α	angle between thrust axis and velocity vector, deg

β angle between radius vector of satellite and radius vector of ferry vehicle, deg or rad (see fig. 1(a))

γ flight-path angle measured between velocity vector and local horizontal, deg (see fig. 1(b))

ϵ elevation angle, measured between initial local horizontal and satellite present position, deg

η angle between radius vector to satellite and line of sight, deg

θ inertial pitch angle, measured between initial local horizontal and longitudinal axis (thrust axis) of ferry vehicle, deg or rad

μ lunar gravitational parameter, ft^3/sec^2

σ angle measured between thrust vector and line of sight, deg

ϕ inertial range angle, measured from initial or launch position of ferry vehicle to present inertial position, deg

ω_s angular velocity of orbiting space station, deg/sec or rad/sec

Subscripts:

BO burnout conditions

o initial conditions

s quantities related to orbiting station

1,2 first and second values of pitch rate

A dot over a symbol indicates a derivative with respect to time.

EQUATIONS OF MOTION

Spacecraft motion was limited to the plane of the orbiting vehicle. The three-degree-of-freedom equations of motions, presented in the appendix, were solved on an electronic analog computer. It was assumed that the vehicle had no automatic damping or automatic control. The pilot closed the control loop and had direct input into the force and moment equations.

The vehicle force equations were written in polar form with the origin at the moon's center (fig. 1), and the moon was assumed to be a homogeneous nonrotating sphere. Vehicle mass and moment of inertia variations with fuel expenditure were taken into account; however, mass changes due to reaction control jets were neglected since this change was negligible compared to mass change due to main engine thrust.

DESCRIPTION OF SIMULATED VEHICLE

The configuration assumed for this investigation is shown in figure 2. The vehicle had a large fixed engine with the thrust axis along the vehicle's axis of symmetry and a maximum thrust to initial mass ratio (T/m_o) of about 24.5 ft/sec^2. The engine was assumed to have a throttleability ratio of about 12 and to be capable of restarts at any time during the mission. Specific impulse I_{sp} was assumed to be 424 seconds. Although these characteristics may not be attainable in practice, it was of interest to investigate pilot and system demands.

Moment control about the pitch axis was assumed to be available from reaction jets located on top the vehicle, and an acceleration command system was assumed to control vehicle attitude.

Coupling was neglected since the control force was small and had no effect on the trajectories. A torque-to-inertia ratio of about 0.65°/sec^2 was used and assumed constant since the center-of-gravity shift and the change of inertia due to fuel expenditure were such that the term l/I_Z was invariant.

SIMULATOR AND ASSOCIATED EQUIPMENT

Cockpit and Controls

Figure 3 shows a general layout of the simulator. Shown on this figure is the pilot's couch, instrument panel, controls, and the projection system used in the study. Figure 4 shows a more detailed view of the cockpit and controls. The couch was inclined to give the pilot the necessary field of view on the curved portion of the screen. Main engine thrust was commanded by use of a controller on the pilot's left (fig. 4), and thrust varied linearly with controller displacement. Pitch attitude was commanded by an on-off type controller located at the pilot's right.

Instrument Display

In selecting a suitable instrument display, certain factors must be considered. Among the most important are what quantities to be displayed, type of display for ease of scanning and quick interpretation, and availability of the quantities displayed. Since the present study was essentially a dual mission - that is, launch from the lunar surface and rendezvous with an orbiting station - it is apparent that the launch phase may require certain displays which may or may not be necessary for the rendezvous maneuver and vice versa. Initial selection of instruments for this investigation was based on what was believed to be the minimum display required to accomplish this particular mission.

In the initial phase of the study the pilot had the following information displayed to him: time, velocity, altitude, flight-path angle, pitch rate, pitch angle, thrust, line-of-sight range, and range rate. However, as the program progressed, a "display evaluation" was made to determine what instruments the pilot was actually using and what displays could be eliminated or replaced by something simpler - simpler in the sense of the generation and availability of the information. As a result, the final display configuration was chosen in which the pilot was given time, altitude, altitude rate, thrust, pitch rate, range, and range rate, as is shown in figures 5 and 6, and was used in the investigation reported herein.

Since rather good resolution was required in some of the displayed quantities and because the quantities varied over a rather wide range, a brief description of the quantities displayed and the special treatment given to certain of these are as follows:

Time: The clock was a standard aircraft-type instrument with sweep second hand and could be read to an accuracy of 1/2 second.

Altitude: The altimeter was calibrated from 0 to 100 nautical miles and could be read to an accuracy of 1 mile.

Thrust: The thrust-level indicator was graduated from 0 to 10 000 pounds and could be read to within 100 pounds thrust.

Pitch rate: The pitch rate was displayed on a sliding-type meter with manual switching so that two scales were available, $\pm 2.0^{\circ}$/sec and $\pm 0.20^{\circ}$/sec full scale, and could be read to an accuracy of approximately 0.05°/sec and approximately 0.005°/sec, respectively.

Range: Line-of-sight range was displayed with manual switching available to obtain a scale factor of 10. The ranges of the meter were from 0 to 500 nautical miles and from 0 to 50 nautical miles. A light was used to indicate the scale in use and the quantity could be read to an accuracy of 5 nautical miles on the high scale and to 1/2 nautical mile on the low scale.

Range rate: Range rate was displayed on a galvanometer-type instrument and manual switching provided three ranges to the pilot: 0 to 10 000 ft/sec, 0 to 1000 ft/sec, and 0 to 100 ft/sec. The three scales could be read to accuracies of 100 ft/sec, 10 ft/sec, and 1 ft/sec, respectively. The proper scale was indicated by a series of lights over the appropriate switch position.

Altitude rate: The altitude-rate instrument had a range of ± 450 ft/sec and could be read to an accuracy of about 10 ft/sec.

In addition to the various instruments, the pilot also had available charts showing the variation of altitude with time for the nominal trajectories which he was to follow. These were used to monitor the progress of the spacecraft during each flight.

Projection System

The projection system, shown in figures 3 and 4, consisted of two projectors and a screen, the screen being the walls that enclosed the simulator. The right-hand projector cast a small spot of light that represented the satellite. A 25-watt point source and a mirror driven by the angle σ (fig. 1(a)) were used to project the image. The left-hand projector presented the lunar horizon and the star background. The star field display was obtained by projecting light through small perforations in a truncated cone. For the lunar surface, a transparency depicting some surface features was made and a slot cut in the cone to accommodate it. The light source was a standard flashlight bulb powered by a small alternating-current power supply.

There was no range cue due to size change in the satellite projection, and the star background presented no particular portion of the celestial sphere since it was not used for guidance but only for initial launch reference and for attitude information during the rendezvous phase. Figure 5 shows the instrument panel, lunar horizon, and star background as seen by the pilot.

TRAJECTORY CONSIDERATIONS AND PILOTING PROCEDURES

Flight Task

The flight task to be simulated was that of launching from the lunar surface and performing a direct rendezvous with a spacecraft which was in a 100-nautical-mile circular orbit. The pilot's tasks were to determine visually lift-off time so that the launch was within the permissible time limit, to follow a prescribed pitch program, and to maintain thrust long enough to insert the vehicle into a coasting orbit which would become tangent to the 100-nautical-mile orbit. At the point of tangency the pilot was to apply thrust in such a manner as to maintain altitude and to perform the rendezvous. The flight trajectory and pertinent phases of the mission are illustrated in figure 7. Runs were terminated when line-of-sight range and range rate were reduced to 1 to 2 miles and 10 to 20 ft/sec, respectively.

Nominal Launch Trajectories

Since the pilot was to control the spacecraft, it was desirable to develop simple guidance procedures for launch. At the same time it was desirable to make the launch economical. On this basis, three gravity-turn launch trajectories were obtained through the use of a high-speed digital computer. These gravity-turn trajectories consisted of a powered phase to an altitude of 20 nautical miles, a coast phase which subtended central angles of 24°, 90°, and 180°, and an apocynthion of 100 nautical miles. The variations of thrust vector orientation with time for these trajectories are shown by the dashed lines in figures 8(a), 8(b), and 8(c) for the coast angles of 24°, 90°, and 180°, respectively. It was found that each of these gravity-turn trajectories could be closely approximated by using linear variations of pitch angle with time as shown by the solid-line curves in figure 8. The trajectories generated by the linear step pitch rate were used as the reference or nominal trajectories in this study. These three nominal trajectories, having coast angles of 24°, 90°, and 180° from booster burnout to apocynthion, required thrust levels corresponding to T/W_o of 0.75, 0.45, and 0.35, respectively. The particular transfer (or coast) angles were chosen because it was felt they covered a fairly wide range of total trip times, presented a reasonable cross section of T/W_o, and gave launch window advantages that will be discussed in a subsequent section. Some of the important parameters associated with each of the three nominal trajectories are presented in table I. Figure 9 gives the time histories of the velocity, altitude, and flight-path angle of these nominal trajectories from launch to insertion into the coasting orbit.

As mentioned earlier, the rendezvous phase began at the end of coast with the ferry vehicle tangent to 100-nautical-mile station orbit. For these nominal trajectories the total fuel expenditure was based on the fuel required to arrive at apocynthion plus the fuel required to impulsively add the necessary velocity change required in the rendezvous. This quantity was used as an assessment of the piloted runs and is referred to hereafter as the ideal fuel required.

Pilot Techniques

There were three basic phases in the mission, the launch, coast, and rendezvous. (See fig. 7.) The procedures which were developed were somewhat different for each phase and for convenience are discussed in order.

Launch phase.- The pilot determined lift-off time by observing the position of the orbiting station against the star background. Having set the throttle at the proper level, depending on the transfer chosen, the procedure was to thrust vertically for 5 seconds then hold the pitch rate at the first nominal value. At the end of a specific time (fig. 8), the pitch rate was adjusted to the second nominal value. Near the end of the launch period the pilot closely monitored the time and altitude displays and adjusted the pitch rate to reach nominal burnout altitude by the end of nominal thrust termination. Because of the limited display, it was felt advisable to terminate the launch at the nominal time due to the sensitivity of apocynthion altitude to launch burnout conditions. For example, an error of about 1 ft/sec in tangential velocity at injection results in an altitude error of about 5000 feet at apocynthion for the 180° transfer trajectory.

Coast phase.- During the coast phase the pilot's task was to compare the meter readings of altitude with those of the nominal ascent trajectory, as a function of time, to obtain some measure of how well he was following the nominal path. If departures from the nominal were noted, the pilot was to wait until near the end of the coast, and then apply thrust radially to adjust altitude to the proper value. The purpose of waiting until the final minutes of this phase to make corrections was due to the fact that near apocynthion, the pilot could determine his attitude more precisely by using the orbiting station as a cue. This general procedure did not provide for accurate control of vehicle separation distance at ferry apocynthion; however, this presented no major problem.

Rendezvous phase.- The rendezvous phase began with the ferry vehicle in front of the orbiting station at about 100 nautical miles altitude with subcircular velocity. The basic procedure was to determine visually thrust direction and with the reaction control jets, control the vehicle attitude so as to maintain altitude while accelerating the ferry to circular velocity and simultaneously reducing ferry to station separation to within 1 to 2 miles. In determining directions, the pilot used his line of sight to the orbiting station as a convenient indication of the local horizontal and local vertical. This was approximately correct since the two vehicles were about at the same altitude with only a few degrees of angular separation. Throughout this phase the pilot monitored ferry-to-station range and range rate along with the altitude and altitude-rate displays.

RESULTS AND DISCUSSION

The results presented in this paper were obtained with the author as the pilot. Several other subjects, including engineers with and without simulator experience, also flew the simulator. In general, the results obtained (after a few practice runs) were comparable to those presented herein.

Launch and Coast Phases

The pilot experienced little difficulty in manually controlling the attitude rate of the vehicle and following closely any of the three nominal launch trajectories. This can be seen in figure 10 which shows typical analog records of the velocity, altitude, flight-path angle and pitch angle against time for the launch phase of the three transfer trajectories. The nominal values of these parameters are indicated in the figure by circles. Readout of the important injection parameters, velocity, altitude, and flight-path angle at launch thrust termination indicated some deviation from the nominal of these quantities; however, the deviations were small and the pilot was able to correct for them during the latter part of the coast phase. The apocynthion altitude and velocities were generally close to the nominal values. The range between the vehicles at apocynthion varied somewhat, depending on the accuracy of launch time, the particular trajectory being flown, and thrust vector angles employed.

Rendezvous Phase

The pilot procedures which have been described for the rendezvous phase had slight variations in technique for the three transfer trajectories; this was due primarily to the differences in closure rates between the target and the ferry vehicle and the range at apocynthion for the three launch trajectories.

24° transfer.- The apocynthion velocity, for the 24° transfer trajectory was about 3800 ft/sec (table I); therefore, the closure rate between the ferry and target vehicles was about 1400 ft/sec. A minimum separation distance of about 5 nautical miles was required at thrust initiation with the assumed maximum acceleration available in order to simultaneously reduce range and range rate to zero. It was found advisable to plan the launch time to permit the ferry vehicle to reach apocynthion with a separation distance somewhat greater than the 5 nautical miles, and use a reduced thrust level for performing the rendezvous. A chart was developed to aid the pilot in selecting an initial thrust level and thrust vector orientation angle as a function of separation distance at apocynthion (fig. 11). The data for the figure were obtained on the basis that the apocynthion velocity differential was initially about 1400 ft/sec, that both vehicles were at the same altitude, and that the thrust level and line of sight from ferry to station would remain constant. Errors at launch thrust termination were small for this transfer trajectory hence changes in separation at apocynthion due to launch time adjustments could easily be determined, since this is simply the product of the target's angular velocity and the launch time increment.

A typical piloted rendezvous portion of the mission for the 24° transfer trajectory is shown in figure 12.

90° and 180° transfers.- The closure rate at apocynthion was considerably lower for the 90° and 180° trajectories (approximately 200 ft/sec and 100 ft/sec, respectively) than in the previous case. Use of the maximum T/W capability of these transfers would require separation distances at apocynthion between ferry and target too small to be practical. Furthermore, because of the longer coast period and higher injection velocity required for these trajectories, small errors at launch thrust termination could cause errors in the estimated range at apocynthion of ±2 to 4 nautical miles. Adjusting the launch time to allow increased range at apocynthion and use of the lowest assumed T/W still required separation distances smaller than the flight procedure would allow. Therefore, a chart similar to the one discussed in the previous section would have been of little value for these trajectories. The following procedure was used for the rendezvous phase for these transfer trajectories. Upon reaching apocynthion, the pilot maintained altitude by applying thrust radially with short bursts of his main engine, taking advantage of his closure rate to reduce the separation distance. This was followed by a tilt-over maneuver and a final thrust to bring range and range rate to the acceptable values.

Early Launch Trajectories

As mentioned in the preceding section, a technique was used where the pilot adjusted the launch time in order to increase separation between the target and ferry vehicles at apocynthion. This techique gave the pilot greater flexibility in all three transfer trajectories without appreciable fuel cost.

24° transfer.- In the 24° transfer trajectory the pilot launched early and took advantage of the increased range at apocynthion to allow more time to monitor the displays and to control the vehicle during the rendezvous maneuver. Looking at figure 11, it can be seen that a 1° lead at launch, which is equivalent to an increment of about 18 nautical miles in range at apocynthion, reduces the acceleration required for the rendezvous by a factor of four and increases the rendezvous phase time proportionally. With a lead angle of about 3°, which is equivalent to a 1-minute early launch, the acceleration level is reduced to less than 1/8 of the nominal on-time launch value and the rendezvous time has been increased to about 3 minutes as compared with about 40 seconds for the on-time launch case.

90° and 180° transfers.- For the 90° and 180° transfer trajectories, the early-launch capability assured that the ferry vehicle would always arrive at apocynthion with reasonable separation distances, regardless of small injection errors at launch thrust termination. Launches of up to 1/2 minute early could be used efficiently and alleviated the launch on-time problems mentioned earlier for these transfer trajectories.

The results of this study, for both on-time and early launches, are presented in figures 13 to 15. Shown in figure 13 are the results for the on-time launch for the three transfer trajectories investigated. The ratio of total fuel used in the simulation m_a to the fuel required to perform an ideal nominal launch and rendezvous m_i is plotted against transfer angle ϕ. It can be seen from the figure that the fuel cost for all three transfer trajectories was within about 3 percent of the ideal value.

Figure 14 is a plot of the fuel ratio m_a/m_i against early launch margin and lead angle. It should be emphasized that the m_i parameter is for an on-time nominal launch and rendezvous. Figure 14(a) shows the results for the 24° transfer trajectory. Launches up to a minute early are within 3 percent of the ideal value ($m_a/m_i = 1.0$) and compare favorably with the on-time launch fuel expenditure. Figures 14(b) and (c) are some results for the 90° and 180° transfer trajectories, respectively. For on-time launches and up to 1° lead angle, the fuel cost for these trajectories generally was also less than 3 percent. Further extension of the launch margin resulted in excessive fuel usage for both the 90° and 180° transfers primarily because of inefficient thrust application using the pulsing technique mentioned earlier and the extremely long rendezvous phase times required for separations at apocynthion comparable with the 24° transfer (over an hour in some cases).

Launch Window

The launch time margin or launch window available is an important operational parameter. Assuming the availability of the three transfer trajectories investigated in this study, there is a nominal launch time for each trajectory which will result in an interception of the orbiting station at apocynthion of the ascent. The relative position of the target and the ferry vehicle at lift-off for the three trajectories is indicated in figure 7 by the ×'s. Of these three trajectories, the earliest nominal launch time is that time associated with the 24° transfer and the latest is the time associated with the 180° transfer; this results in a launch window of about 4 minutes.

An operational procedure then would be to have the pilot prepare to launch the vehicle at the time associated with the 24° transfer trajectory. If for some reason launch is delayed, he may wait about 2 minutes and select the 90° transfer. If he is delayed even further, he can wait about 2 more minutes and select the 180° transfer.

In figure 15, the fuel ratio m_a/m_i for the three transfer trajectories is plotted against increment in launch time referred to the nominal (on-time) launch for the 24° transfer trajectory. Also included in this figure and indicated by the triangles are some typical results of the early launch for the 24° transfer. There is about a 4-minute window between the earliest nominal launch (24° transfer) and the latest nominal launch time (180° transfer). Furthermore, with the availability of the three assumed transfer trajectories and using the early launch capability of the 24° transfer, a launch window of about 5 minutes can be realized.

CONCLUSIONS

A fixed-base simulator study has been made to determine the ability of a pilot to launch from the lunar surface and rendezvous with an orbiting station using relatively few instruments and certain visual cues. The pilot's task was to visually determine his launch time by observing the satellite against the star background, to command vehicle attitude and main engine thrust so as to follow a predetermined launch trajectory from lift-off to station altitude, and, finally, to rendezvous with the orbiting station. Three different nominal ascent trajectories requiring three different pitch programs and three different cut-off times were chosen. These trajectories correspond to transfer angles from injection to an apocynthion altitude of 100 n. mi. of 24°, 90°, and 180° with initial thrust-weight ratios of 0.75, 0.45, and 0.35, respectively.

The study restricted the pilot to motions in a plane. The vehicle was assumed to have no automatic damping or automatic control. Moment control was about the pitch axis, and an acceleration command system was used to control vehicle attitude. Results of this investigation have led to the following conclusions:

1. The pilot could accurately determine lift-off time by observing the position of the orbiting station against the star background.

2. With only a brief amount of training and limited display information, the pilot could follow closely any of the three nominal launch trajectories by commanding vehicle attitude and main engine thrust.

3. By using the line of sight to the orbiting station as an indication of the local horizontal and local vertical, the pilot could control vehicle attitude so as to perform successfully and efficiently the rendezvous maneuver.

4. Assuming the availability of the three transfer trajectories, the pilot had at his disposal a launch margin of about 4 minutes. The early launch capability extended the launch window to approximately 5 minutes.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Station, Hampton, Va., January 14, 1965.

APPENDIX

EQUATIONS OF MOTION

The three-degree-of-freedom equations of motion used in this piloted lunar take-off and rendezvous simulation are written in polar form, referenced to the center of the moon, as follows (fig. 1(a)):

Vehicle force and moment equations:

$$\ddot{r} = r\dot{\phi}^2 - \frac{\mu}{r^2} + \frac{T}{m}\sin(\phi + \theta)$$

$$\ddot{\phi} = -\frac{2\dot{r}\dot{\phi}}{r} + \frac{T}{mr}\cos(\phi + \theta)$$

$$\ddot{\theta} = \frac{Fl}{I_Z}$$

Auxiliary equations (figs. 1(a) and (b)):

$$\dot{m} = \frac{-T}{g_e I_{sp}}$$

$$\alpha = \theta + \phi - \gamma$$

$$\gamma = \tan^{-1}\left(\frac{\dot{r}}{r\dot{\phi}}\right)$$

$$V = \left(\dot{r}^2 + r^2\dot{\phi}^2\right)^{1/2}$$

$$\dot{h} = \dot{r}$$

$$\dot{\beta} = \dot{\phi} - \omega_s$$

$$R = \left(r_s{}^2 + r^2 - 2rr_s\cos\beta\right)^{1/2}$$

$$\dot{R} = \frac{r\dot{r} - r_s(\dot{r}\cos\beta - r\dot{\beta}\sin\beta)}{R}$$

APPENDIX

$$\eta = \tan^{-1} \frac{(r \sin \beta)}{r_s - r \cos \beta}$$

$$\sigma = \eta + 90 + \beta - (\phi + \theta)$$

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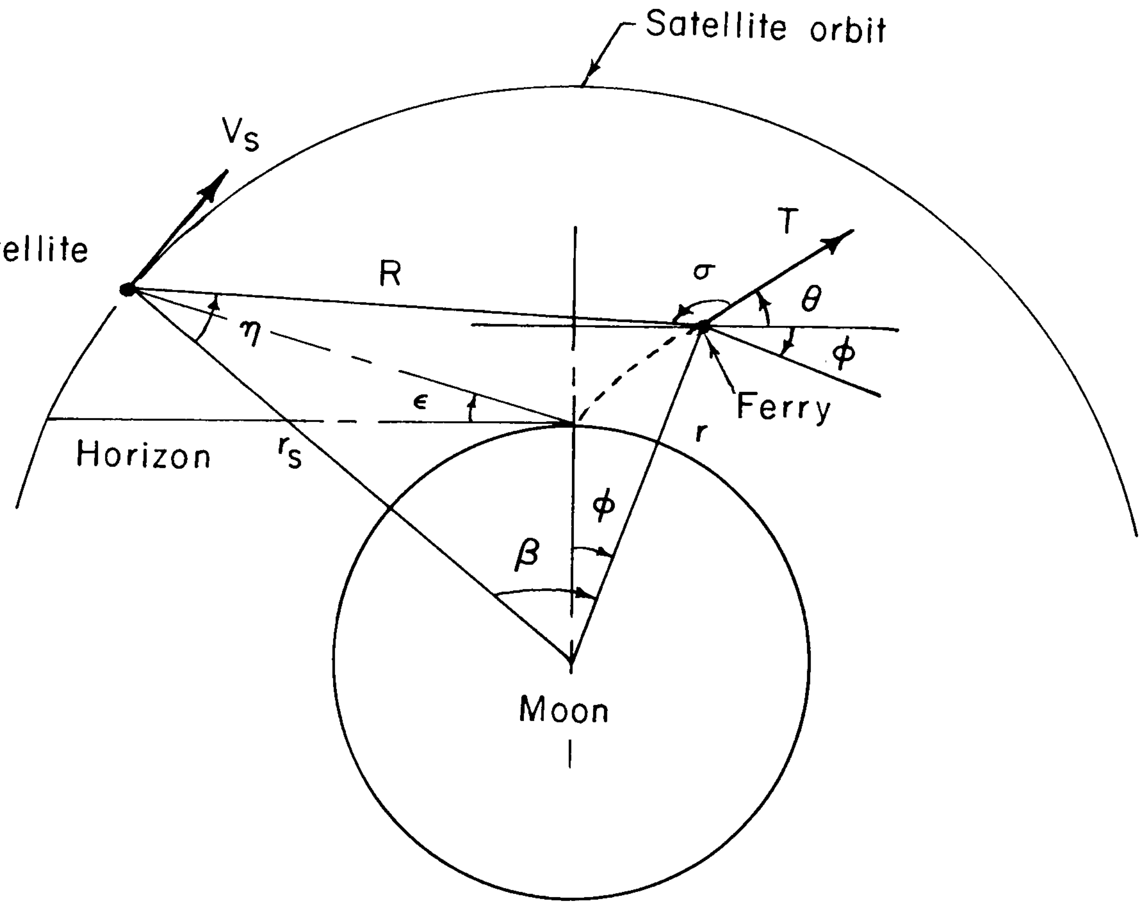
TABLE I.- SUMMARY OF IMPORTANT PARAMETERS AND CONSTANTS

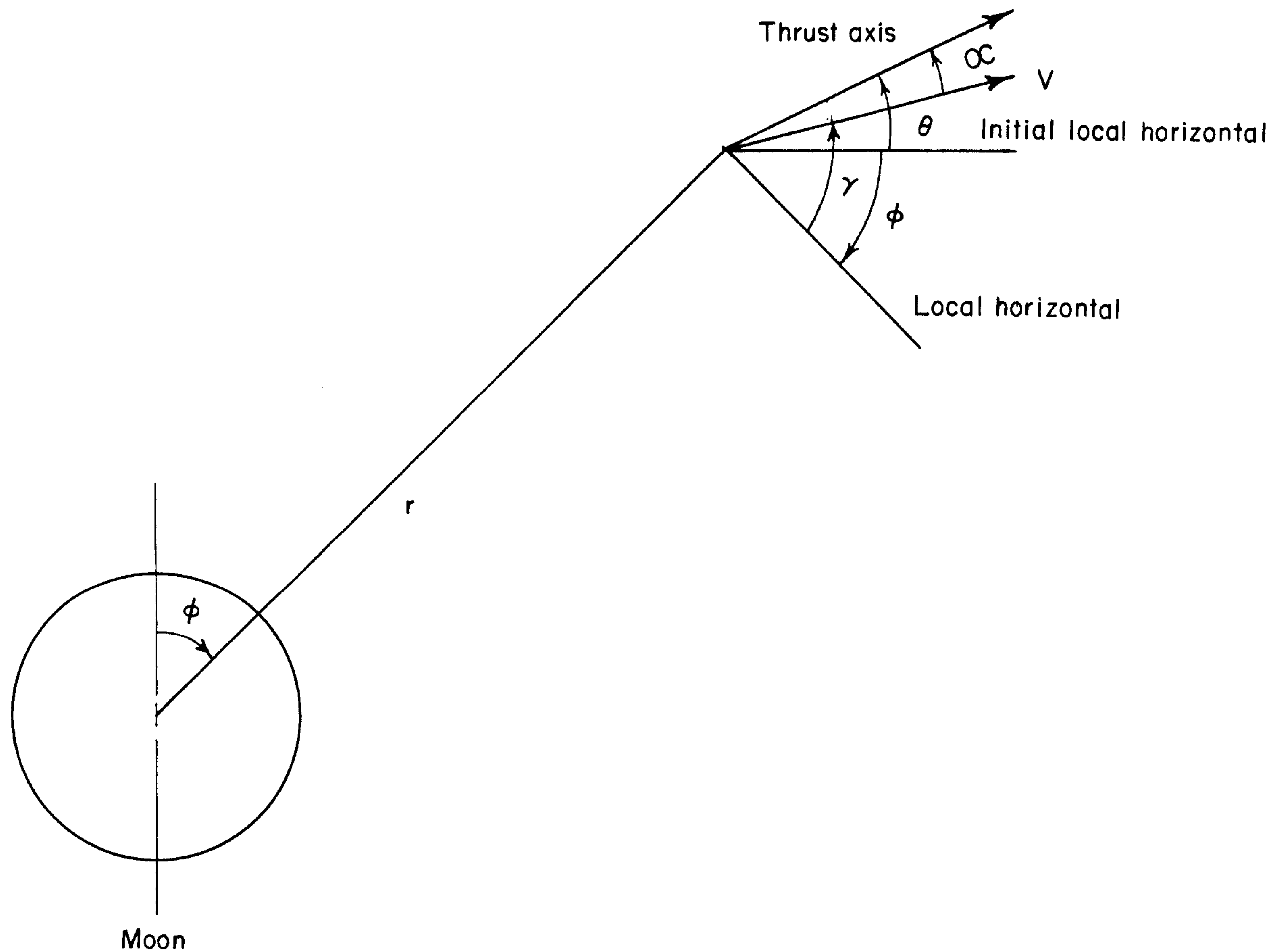
FOR THE THREE NOMINAL TRANSFER TRAJECTORIES

Parameter	Transfer angle of -		
	24°	90°	180°
h, injection, n. mi.	24.5	21.0	19.9
h, apocynthion, n. mi.	100	100	100
I_{sp}, lb-sec/lb	424	424	424
m_o, slugs	336	336	336
m_{BO}, slugs	233	216	212
R_o, n. mi.	240	150	105
T/m_o, ft/sec^2	24.4	14.46	10.96
t_{BO}, sec	171	341	460
t_{coast}, sec	660	1820	3500
Time satellite is in view from $\epsilon = 0$ to ϵ at launch, min	4.4	6.45	8.2
V, injection, ft/sec	4317	5455	5555
V, apocynthion, ft/sec	3791	5029	5127
V_s, 100 n. mi. orbit, ft/sec	5230	5230	5230
γ, injection, deg	18.5	4.2	0.4
$\dot{\theta}_o$, deg/sec	0	0	0
$\dot{\theta}_1$, deg/sec	-1.45	-1.39	-0.729
$\dot{\theta}_2$, deg/sec	-0.037	-0.11	-0.126
$\ddot{\theta}$, deg/sec^2	0.65	0.65	0.65
ϵ, on-time launch, deg	17.5	36	70



(a) Reference-axis system and relationships between station and ferry.

Figure 1.- Reference-axis system.



(b) Angular relationships relative to thrust axis.

Figure 1.- Concluded.

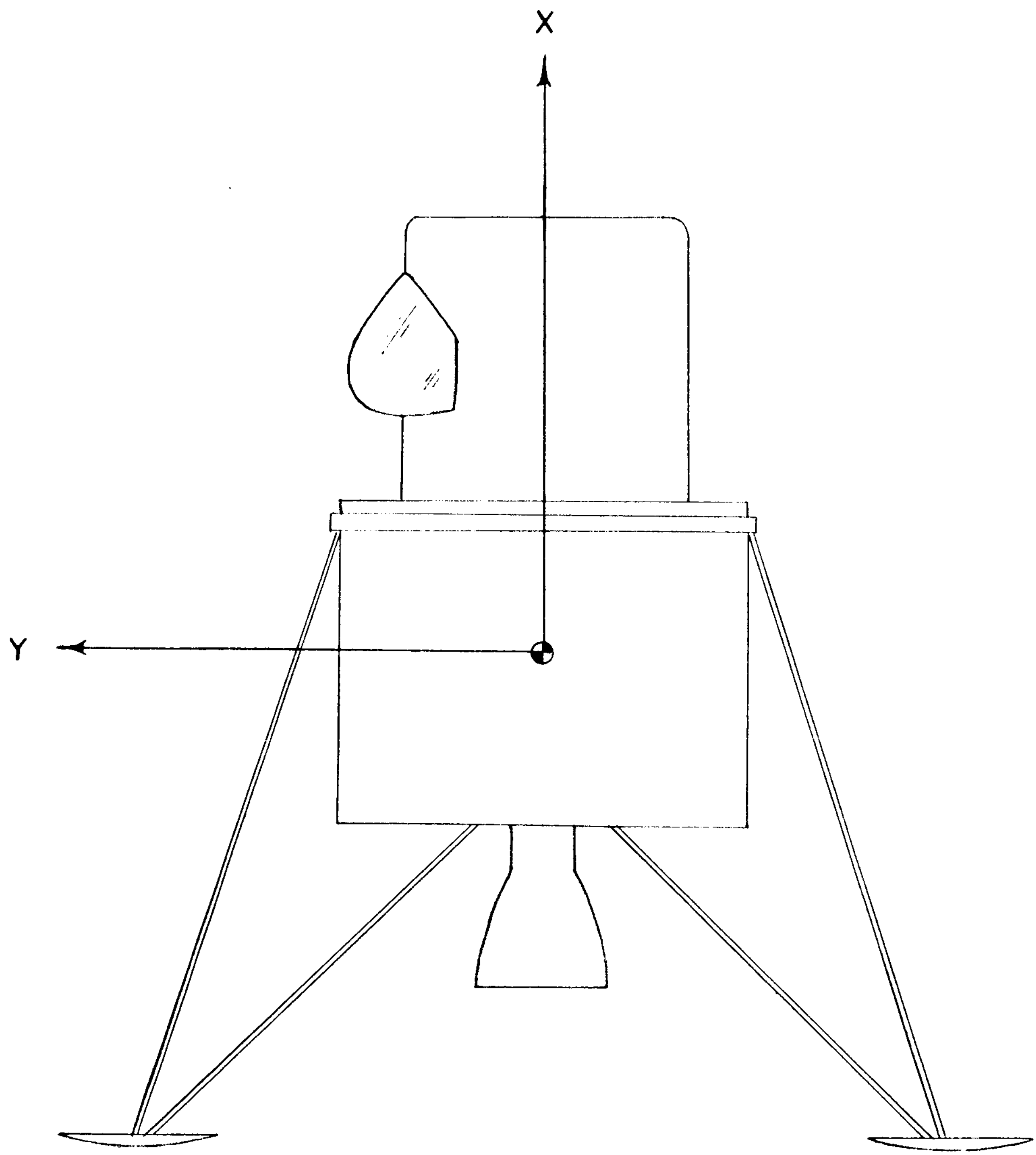


Figure 2.- Sketch of vehicle assumed in simulation.

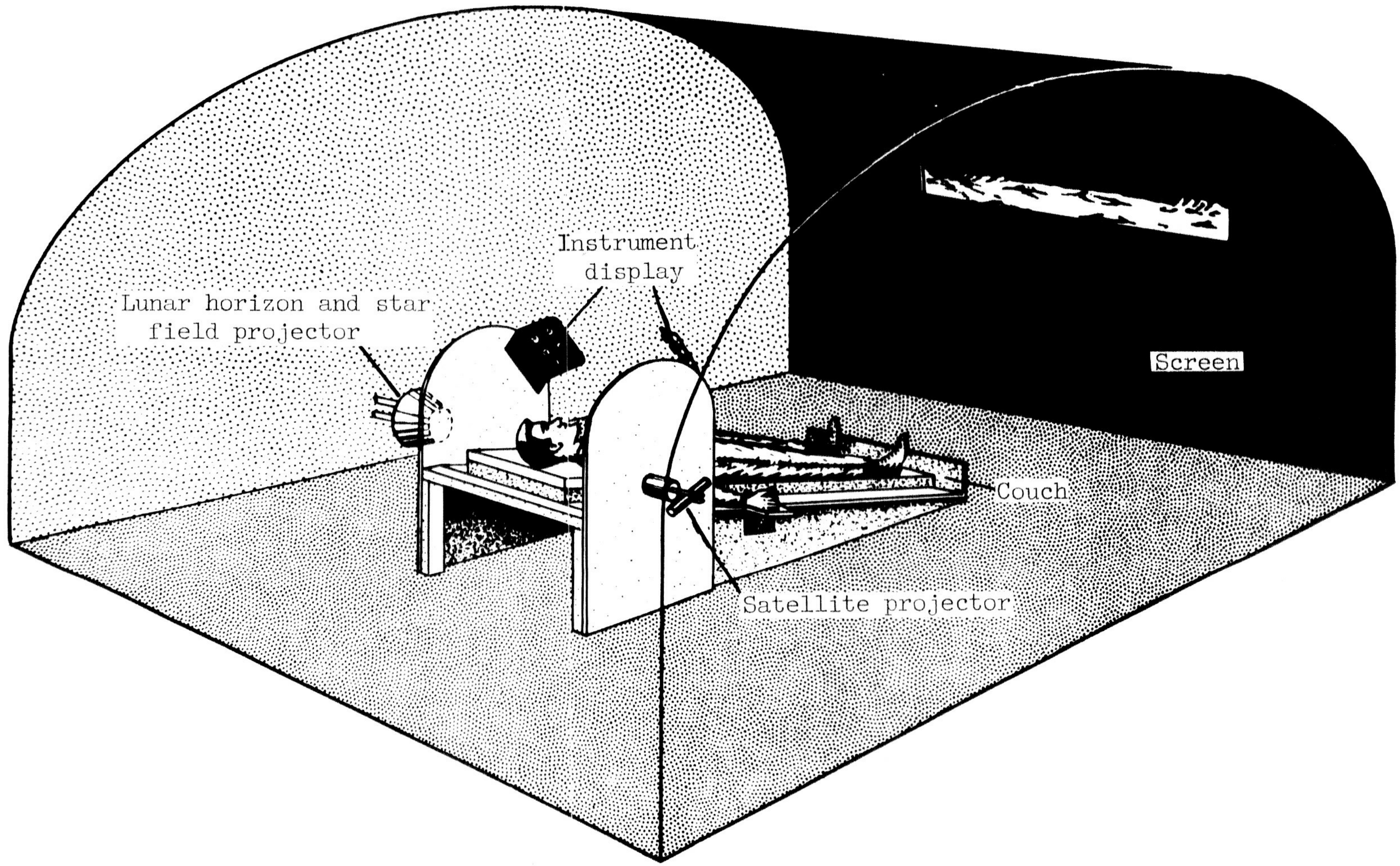


Figure 3.- Artist's sketch of lunar take-off and rendezvous simulator.

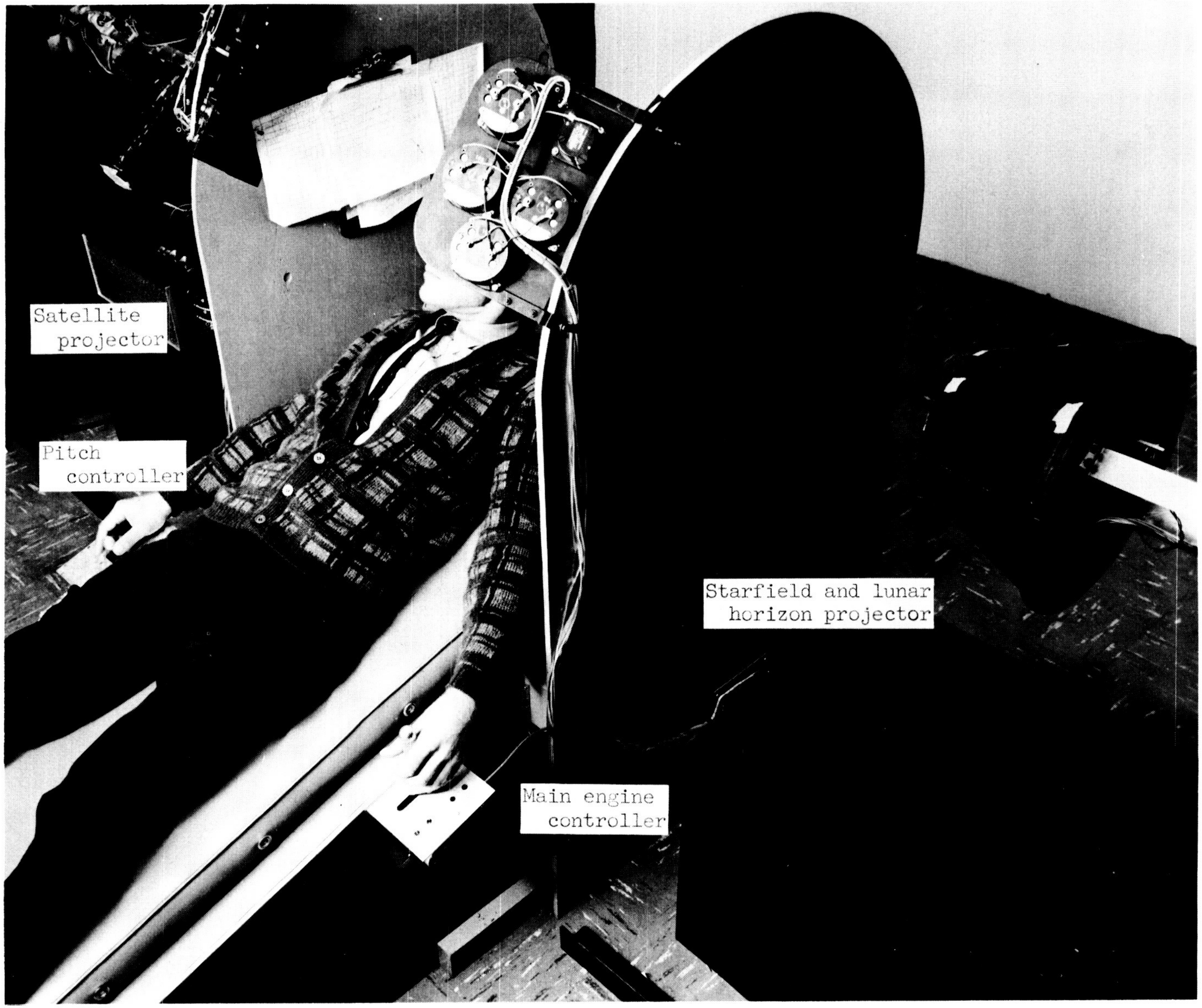


Figure 4.- Photograph of simulator showing some of peripheral equipment. L-62-169.1

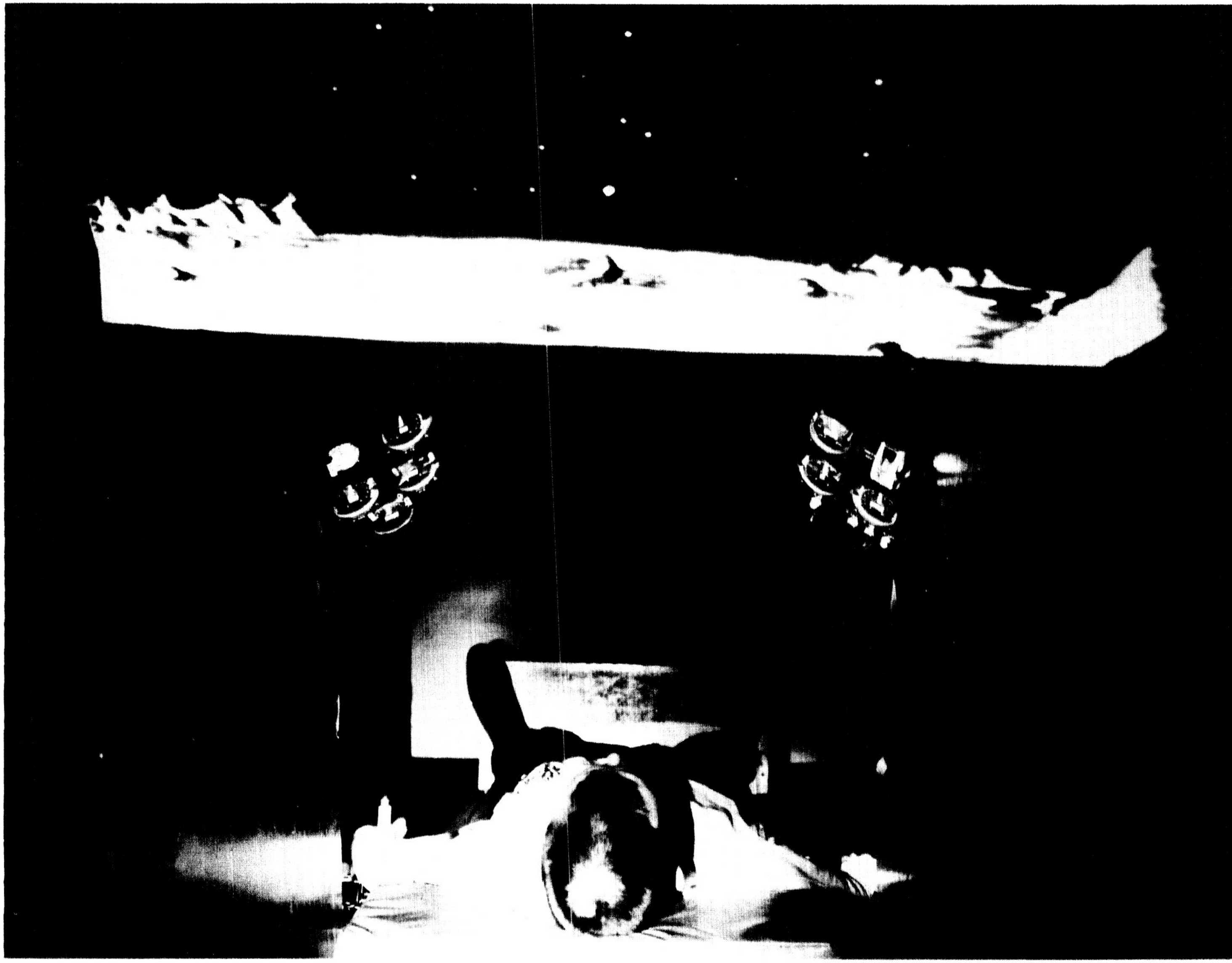


Figure 5.- Photograph of lunar horizon, star background, and satellite as seen by pilot. L-63-1500

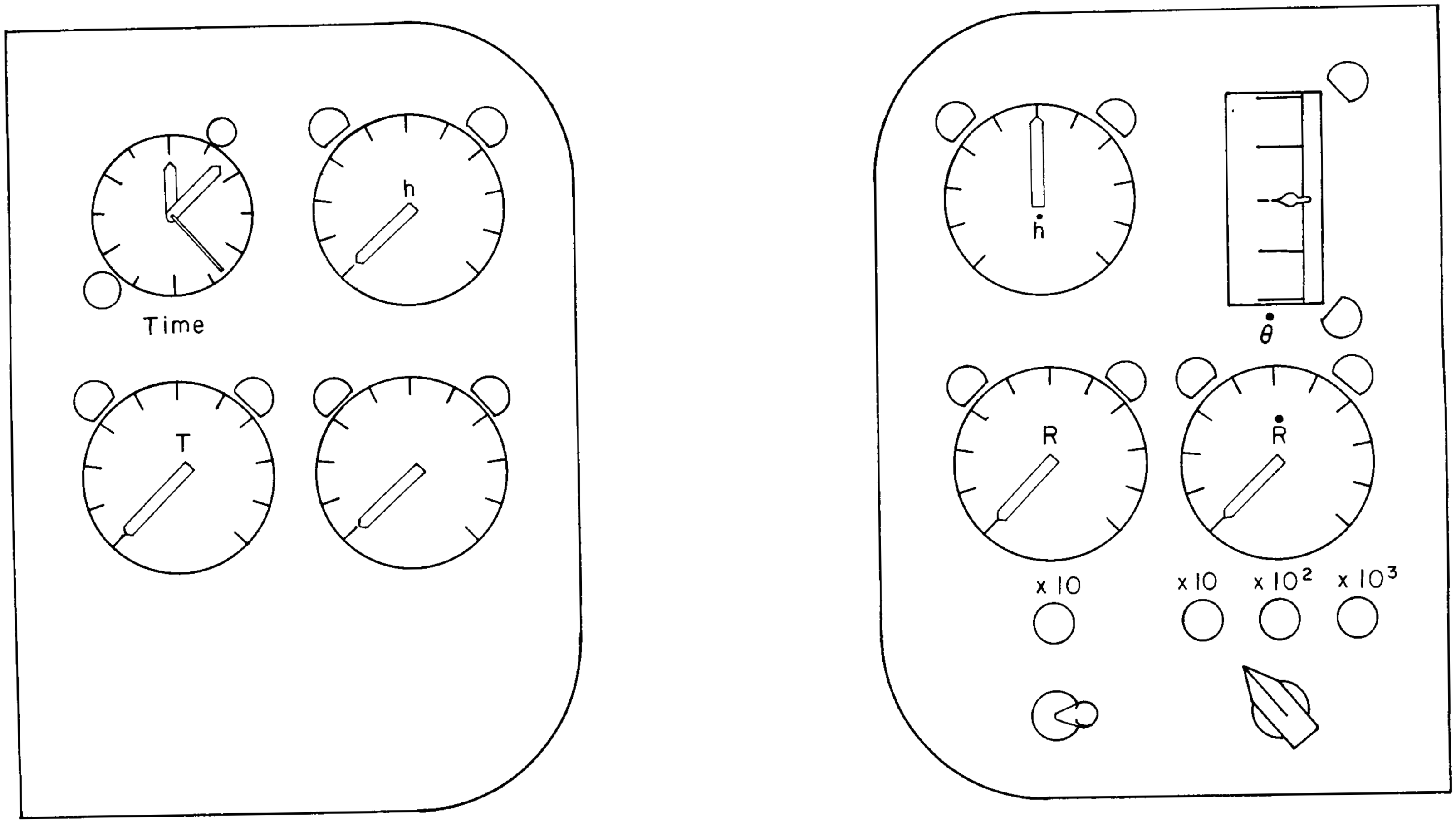


Figure 6.- Sketch of instrument panel and display layout.

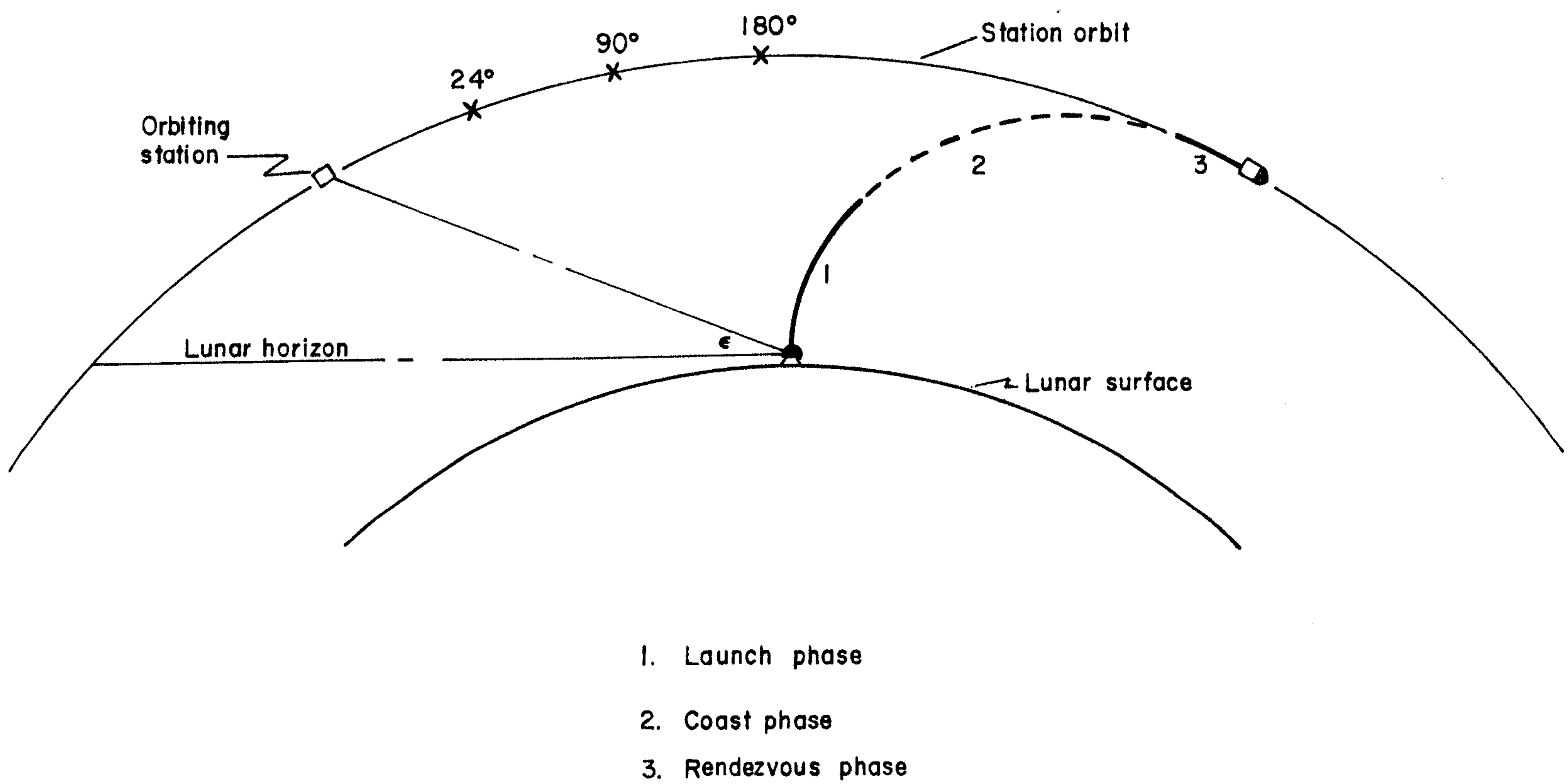
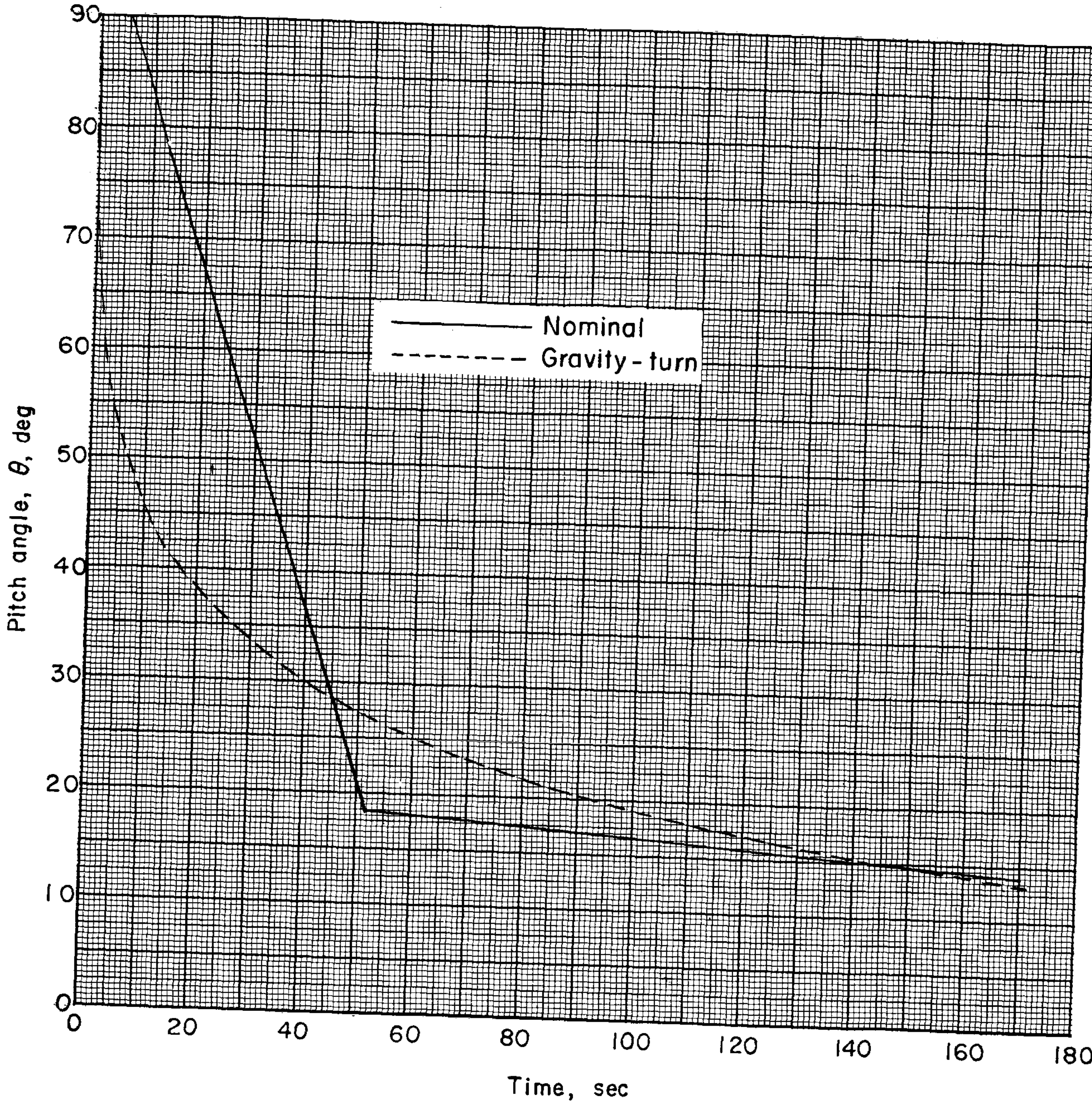
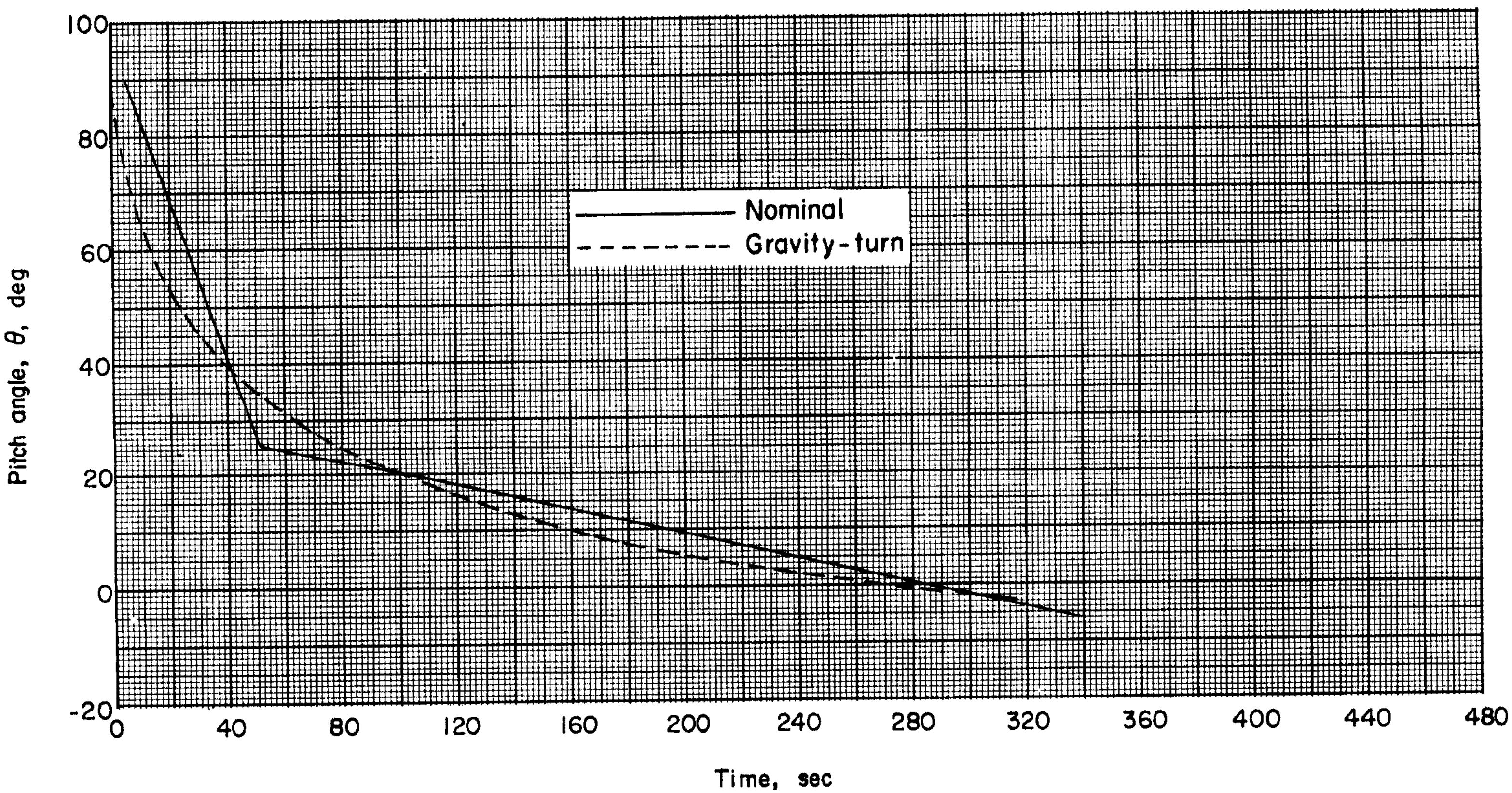


Figure 7.- Sketch of nominal flight trajectory and basic phases of mission.



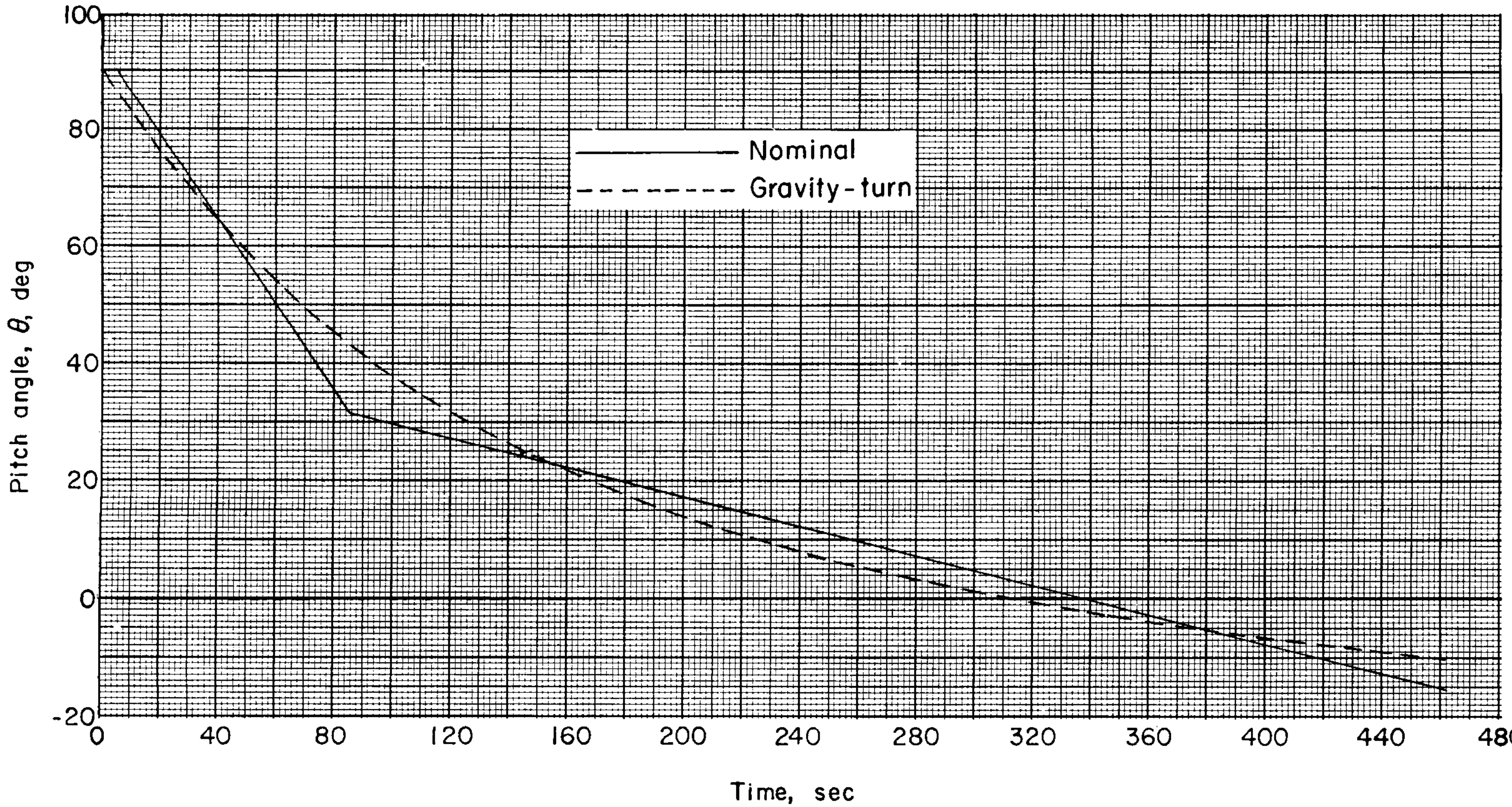
(a) 24° transfer.

Figure 8.- Comparison of powered ascent pitch angle variation with time for gravity-turn and nominal trajectories.



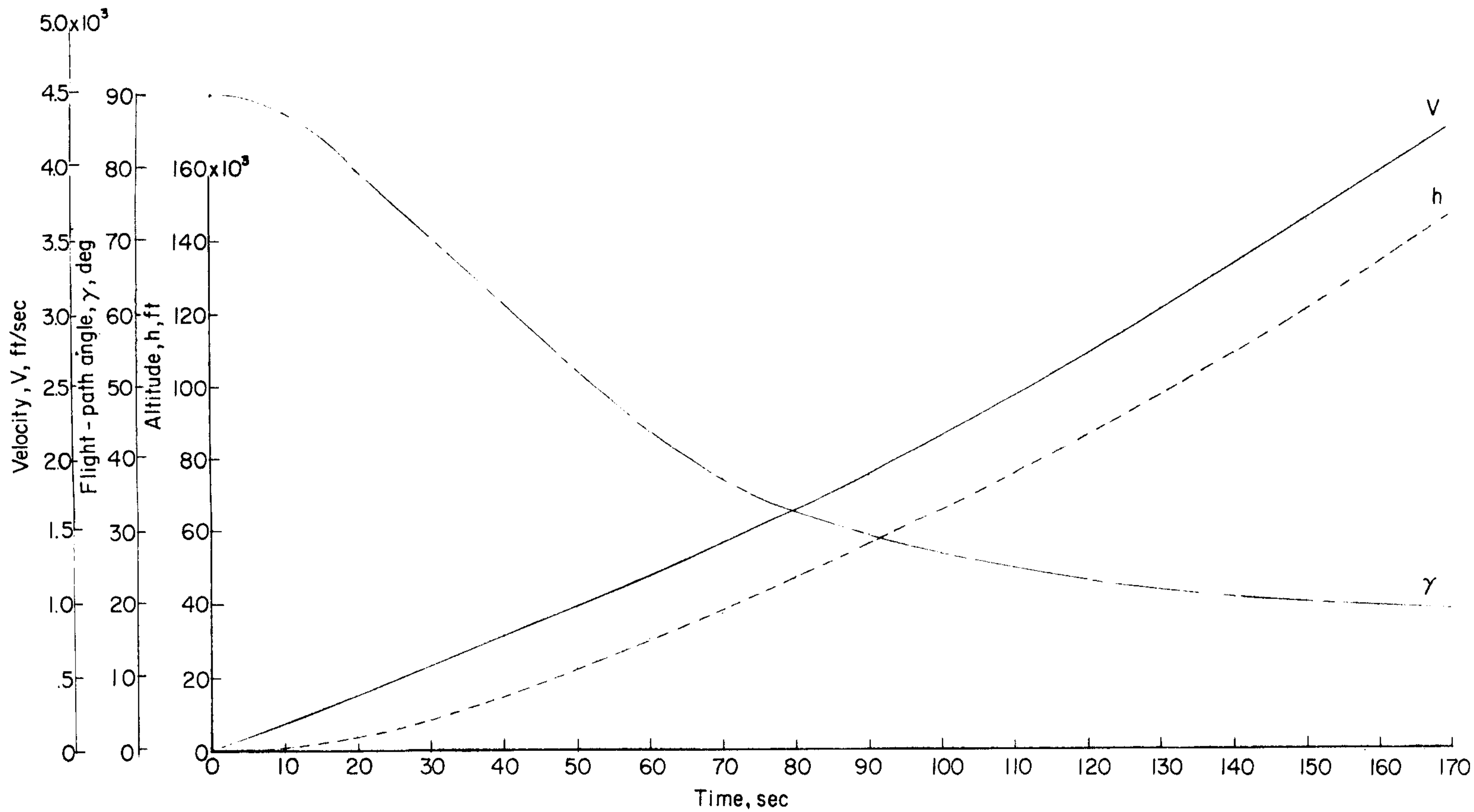
(b) 90° transfer.

Figure 8.- Continued.



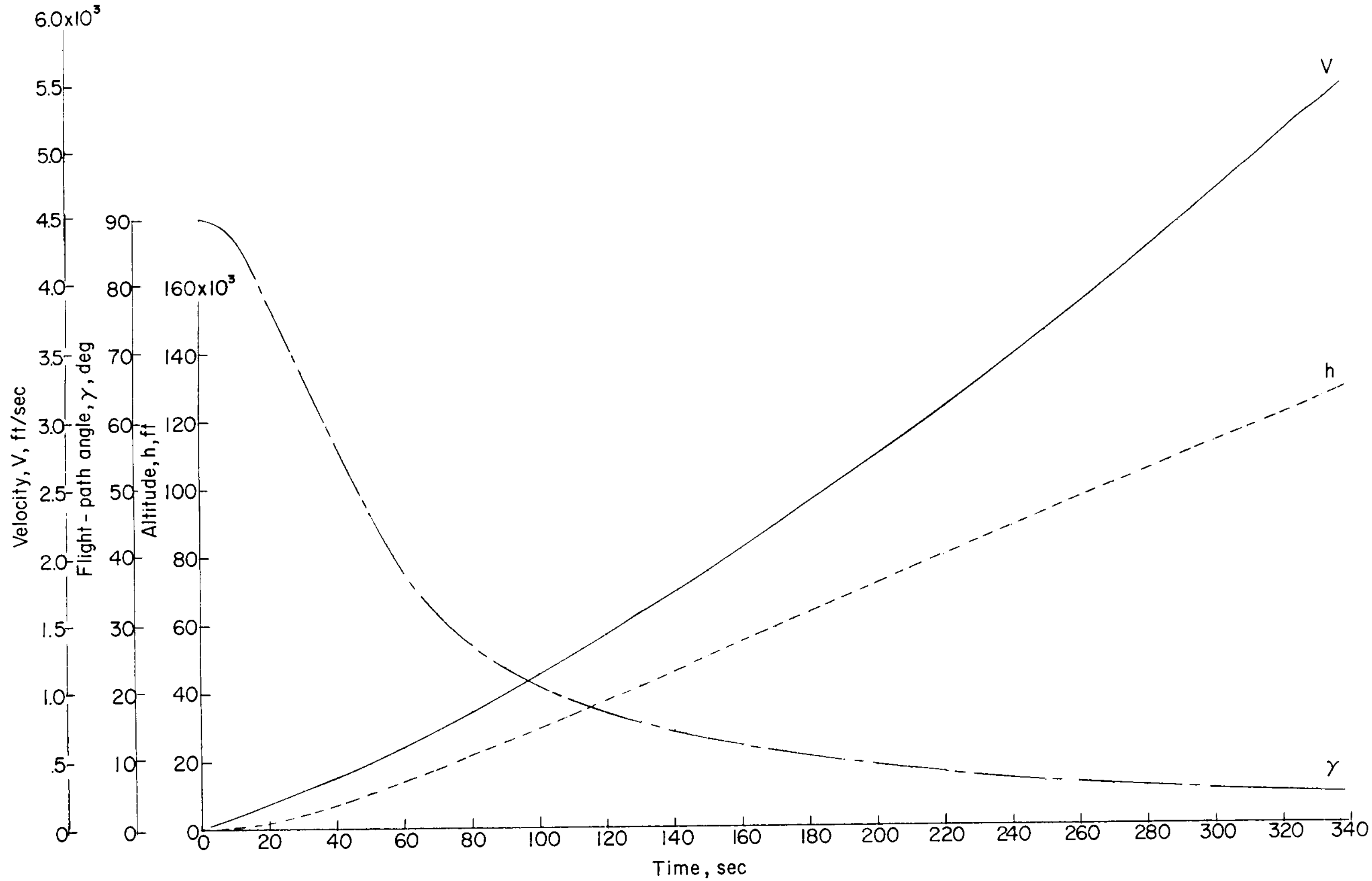
(c) 180° transfer.

Figure 8.- Concluded.



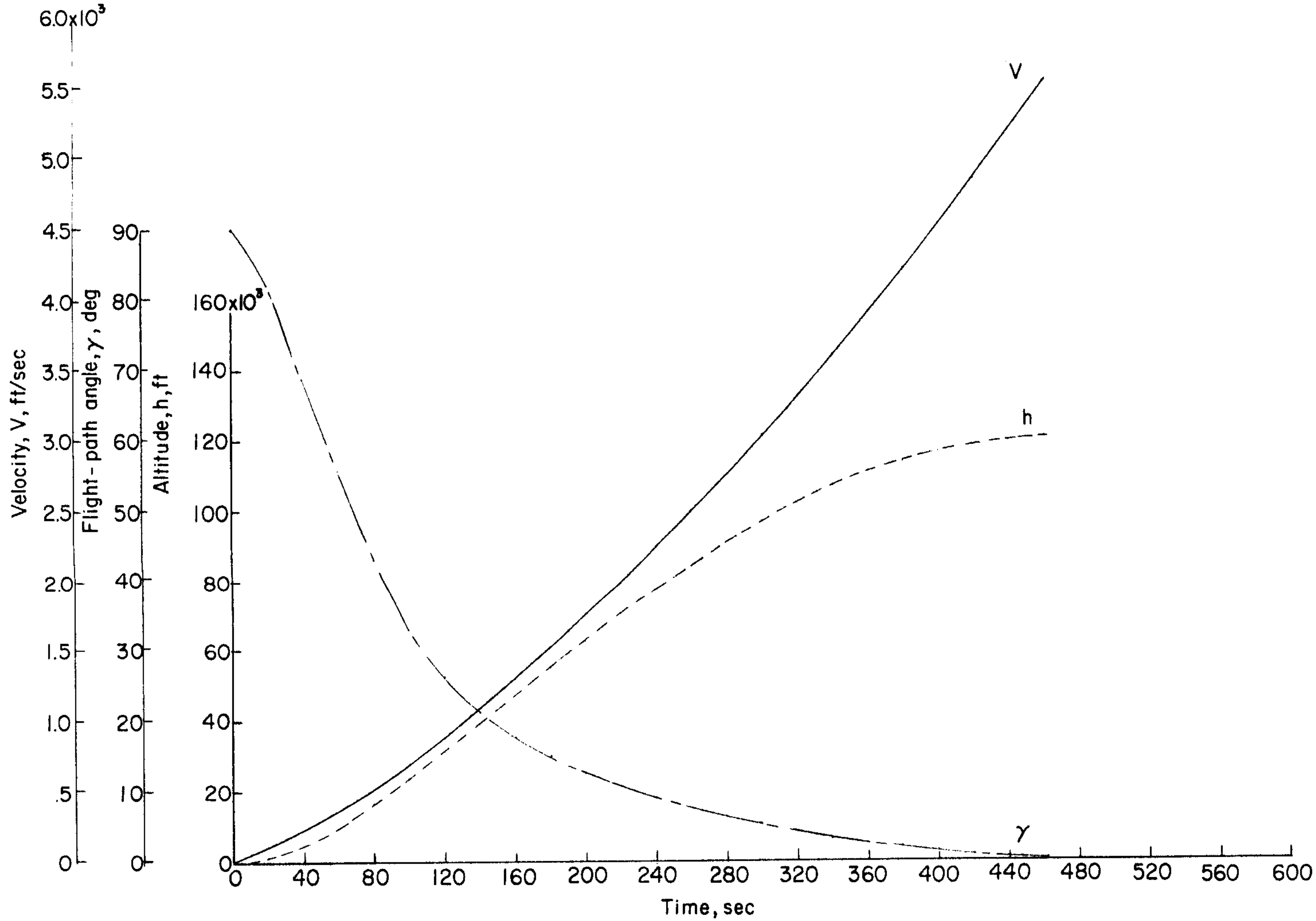
(a) 24° transfer.

Figure 9.- Inertial velocity, flight-path angle, and altitude time histories for nominal ascent trajectories.



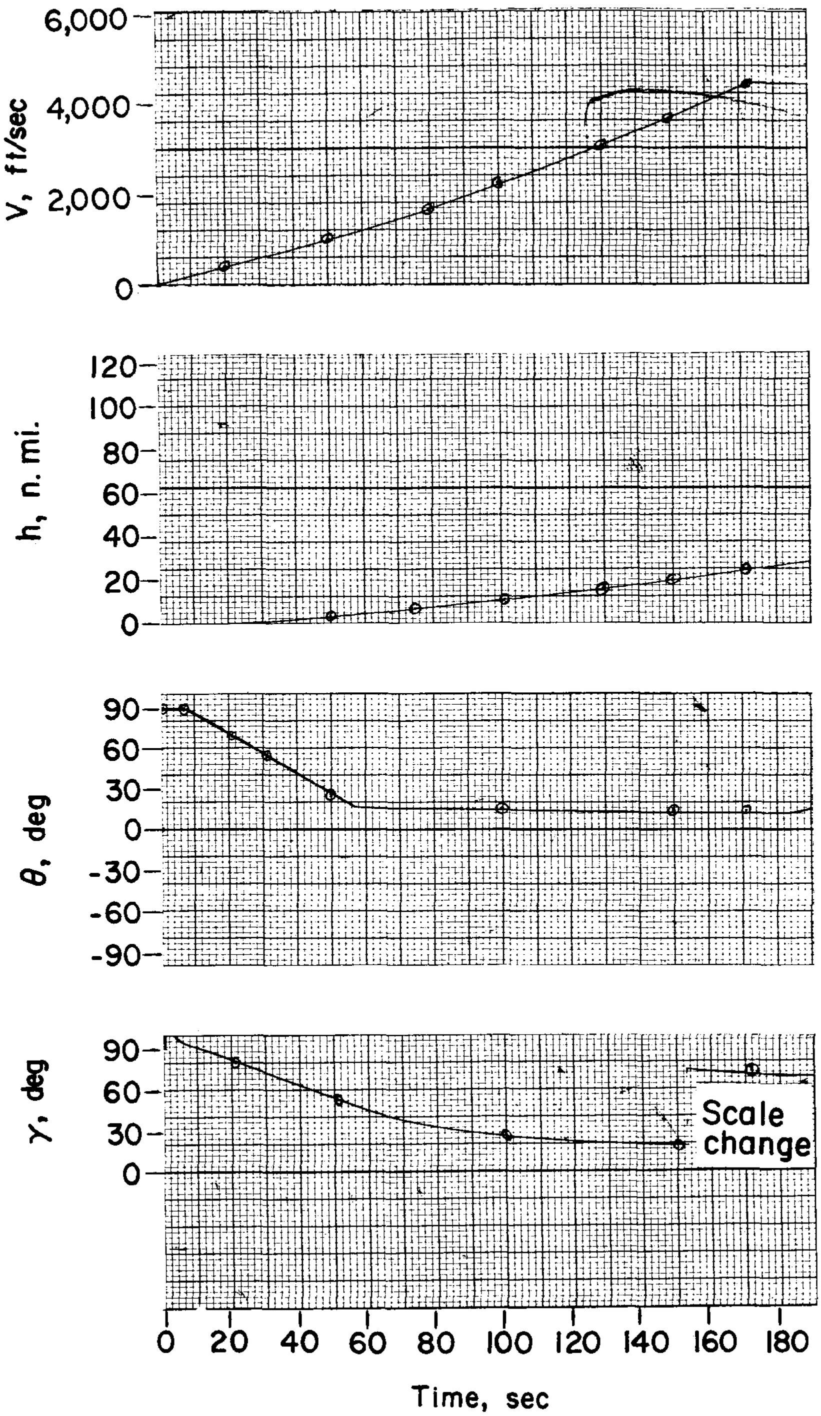
(b) 90° transfer.

Figure 9.- Continued.



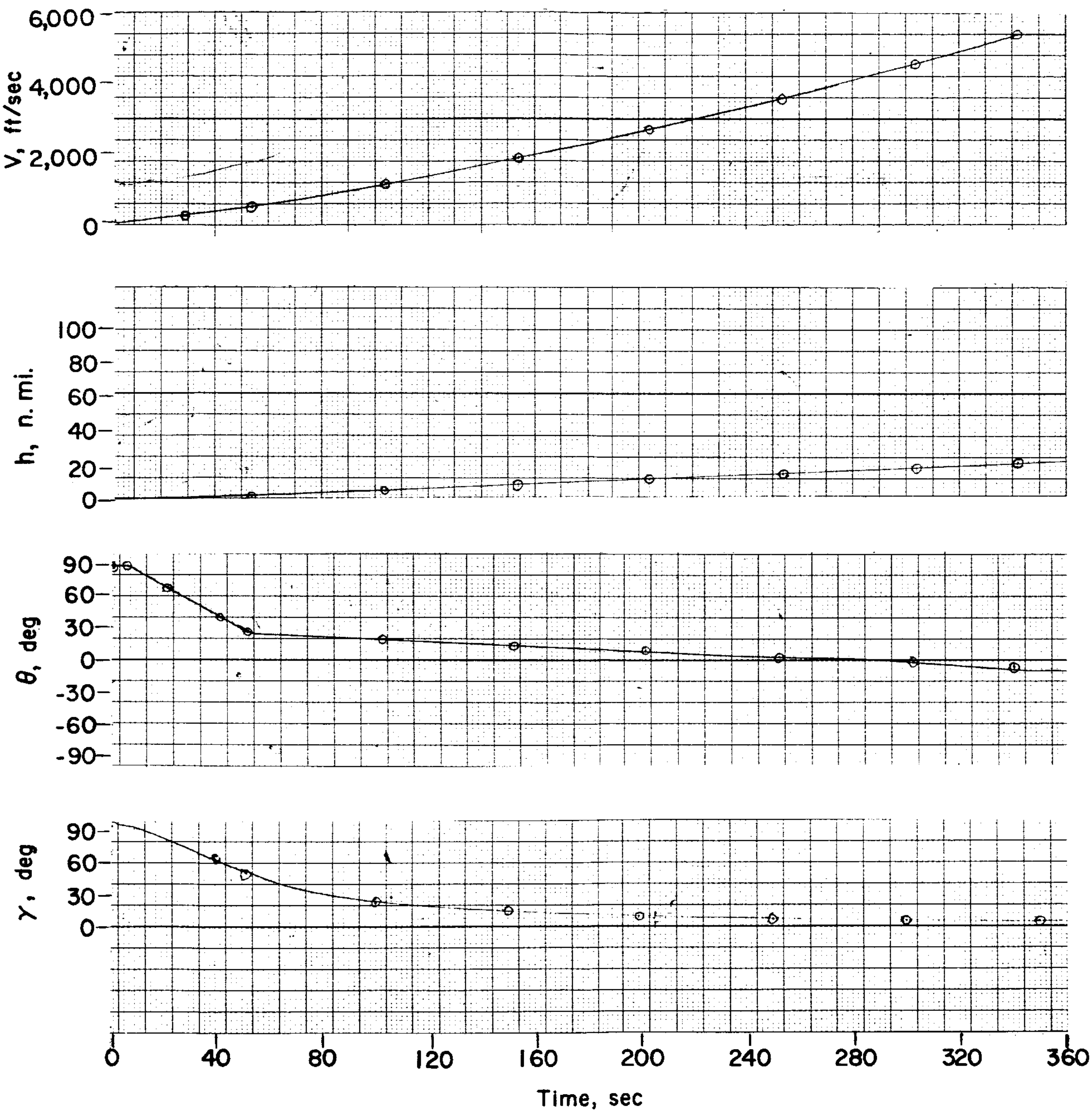
(c) 180° transfer.

Figure 9.- Concluded.



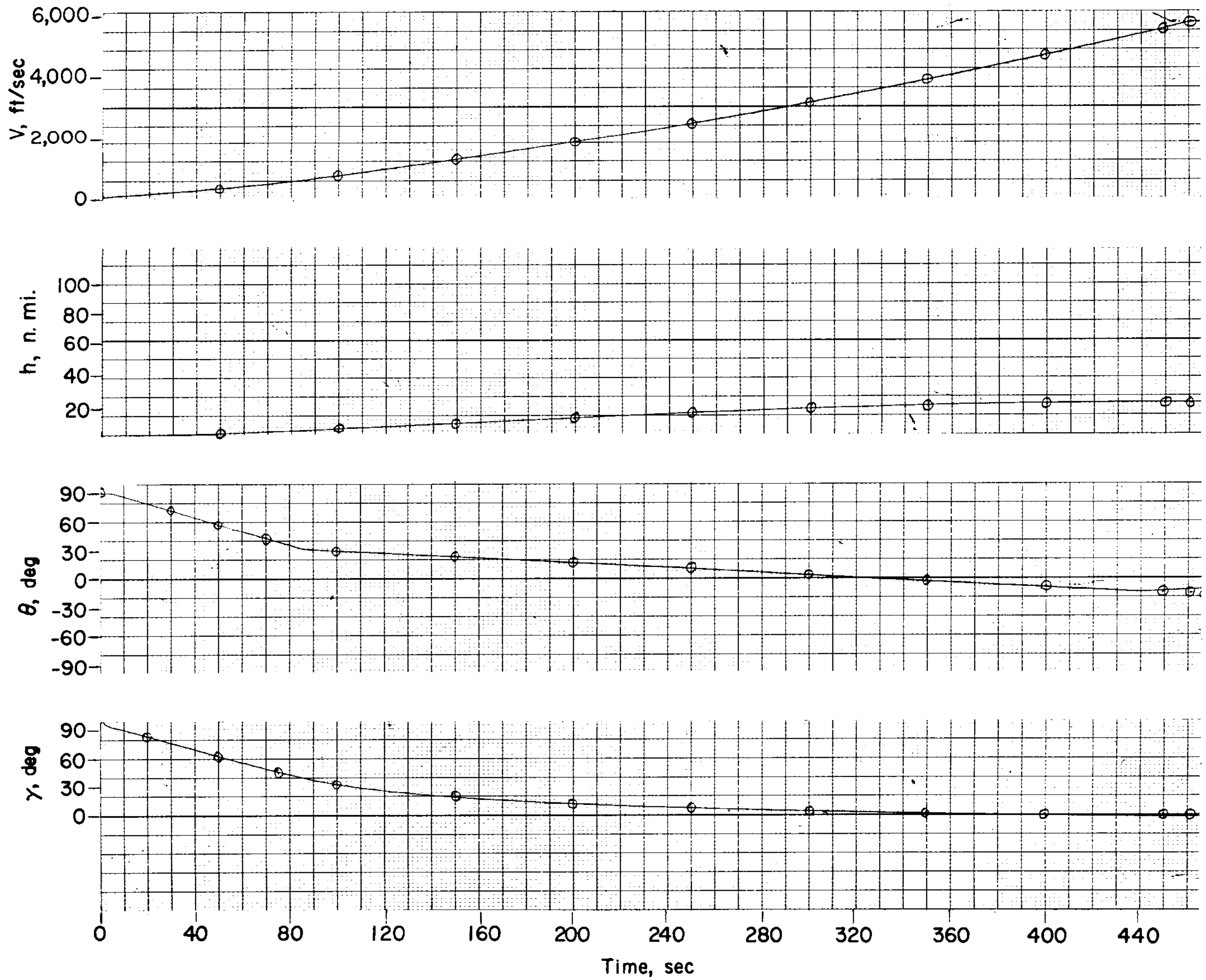
(a) 24° transfer.

Figure 10.- Typical time history of a piloted powered ascent phase. Circles indicate nominal values.



(b) 90° transfer.

Figure 10.- Continued.



(c) 180° transfer.

Figure 10.- Concluded.

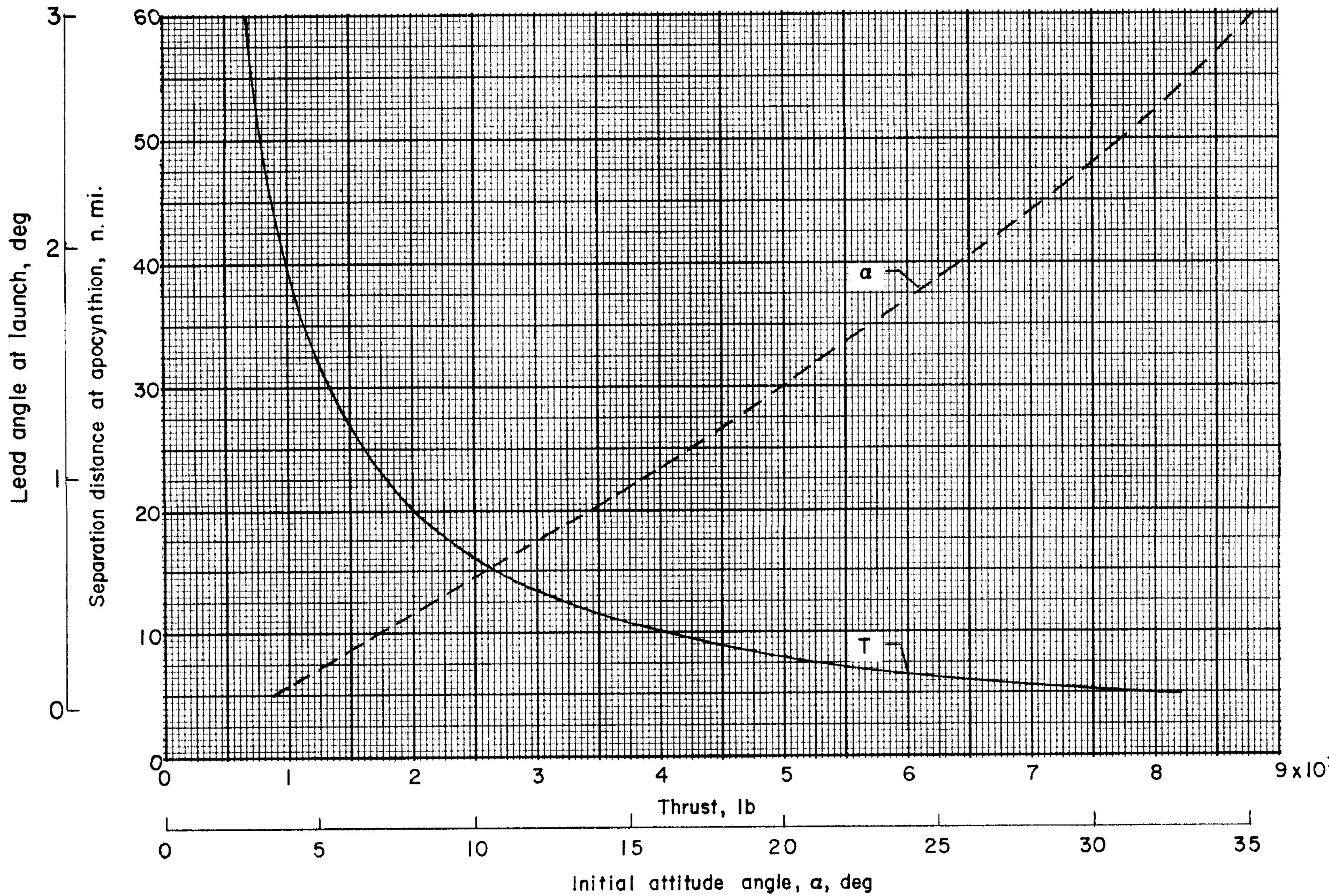


Figure 11.- Relationship of initial thrust and initial thrust vector orientation with separation between two vehicles at apocynthion.

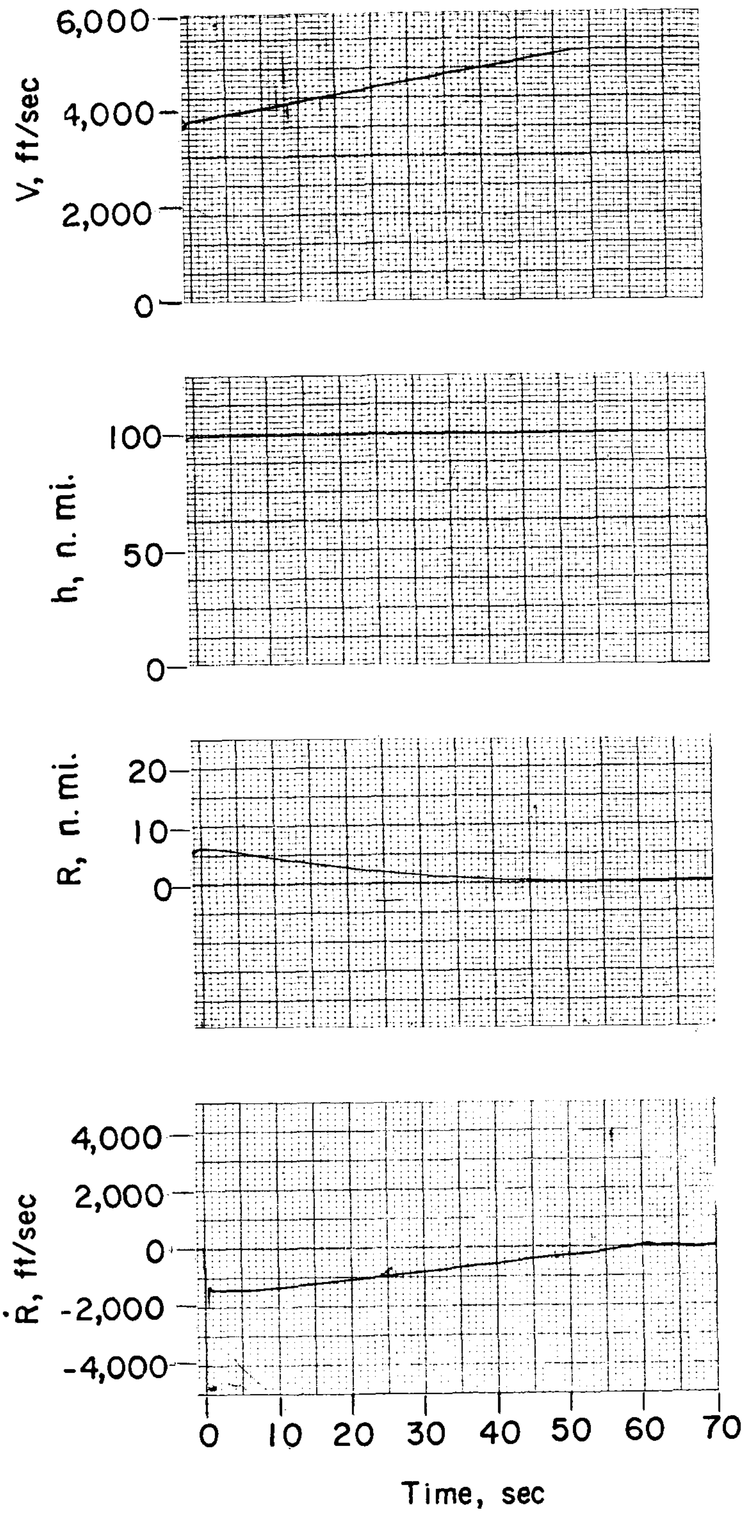


Figure 12.- Time history of a typical piloted rendezvous for 24° transfer trajectory.

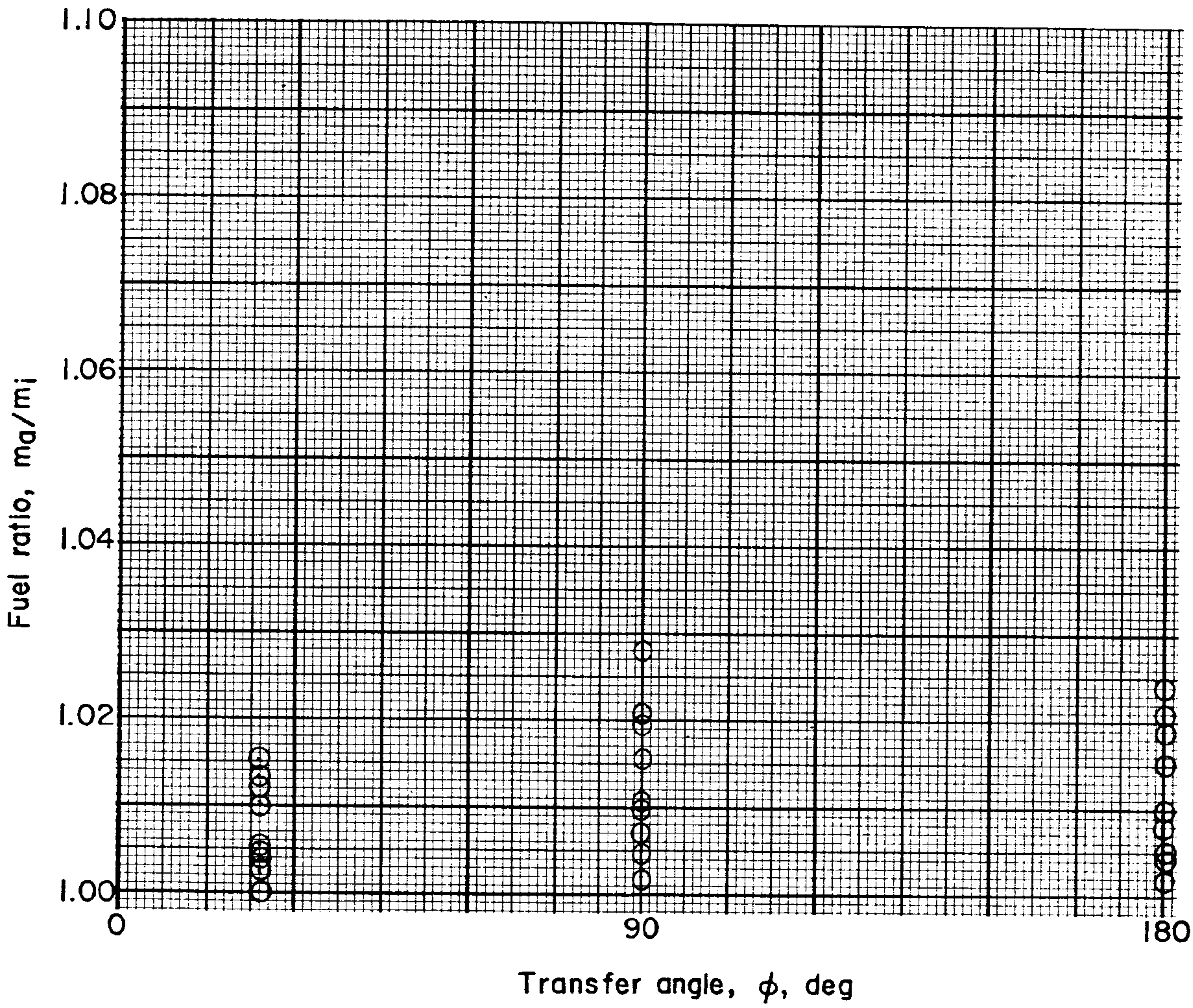
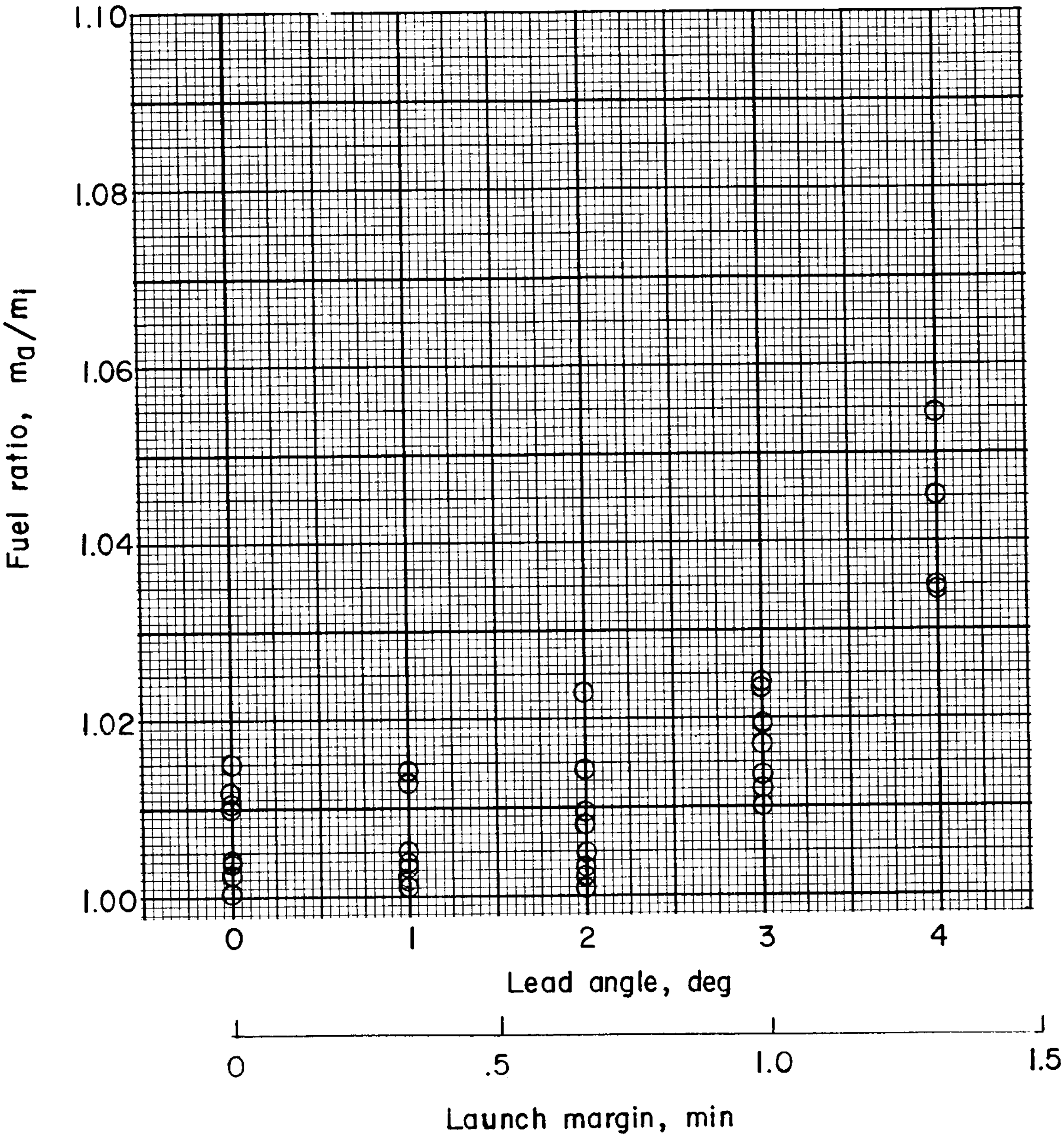
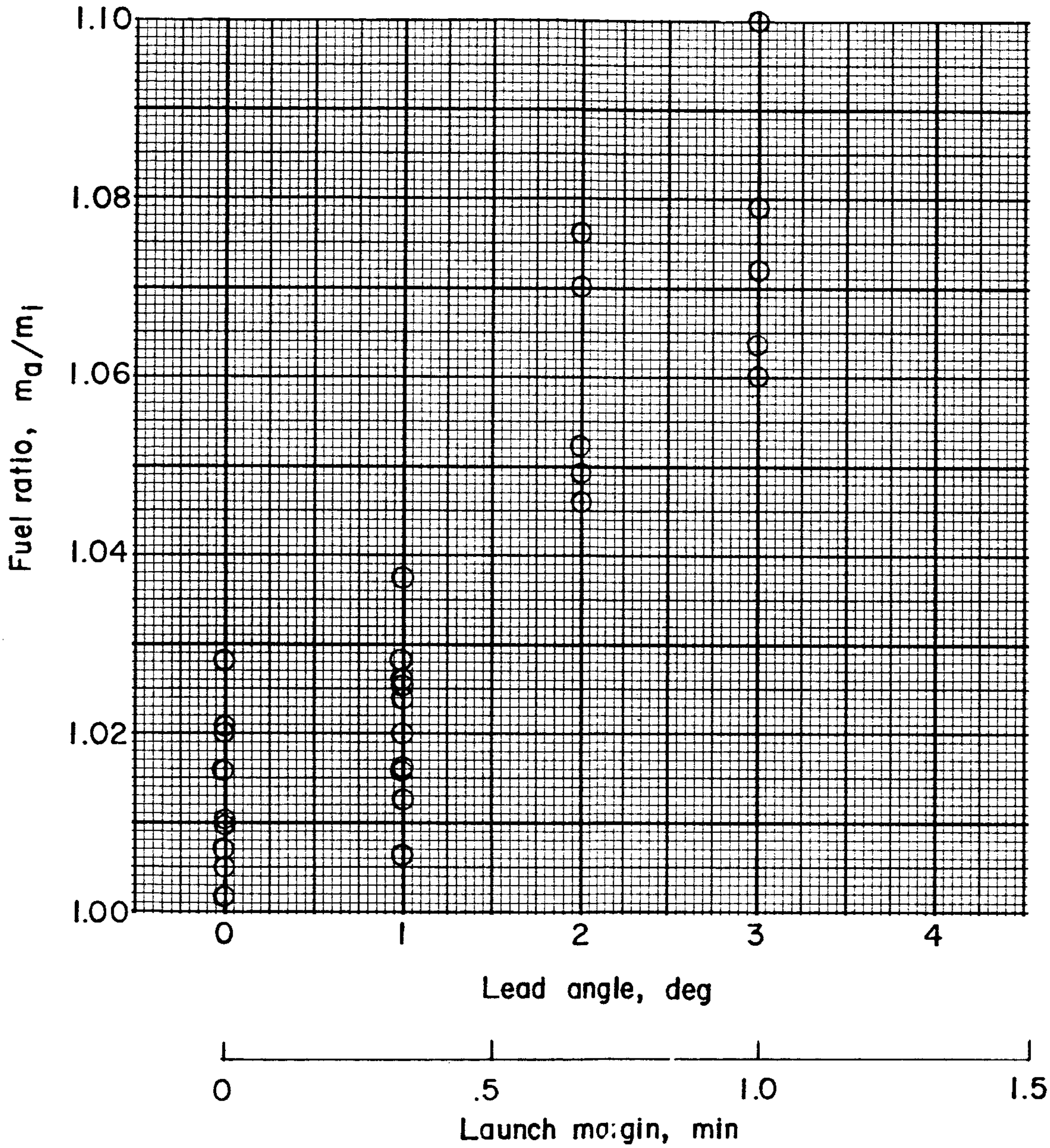


Figure 13.- Results of piloted on-time launch and rendezvous for three transfer trajectories.



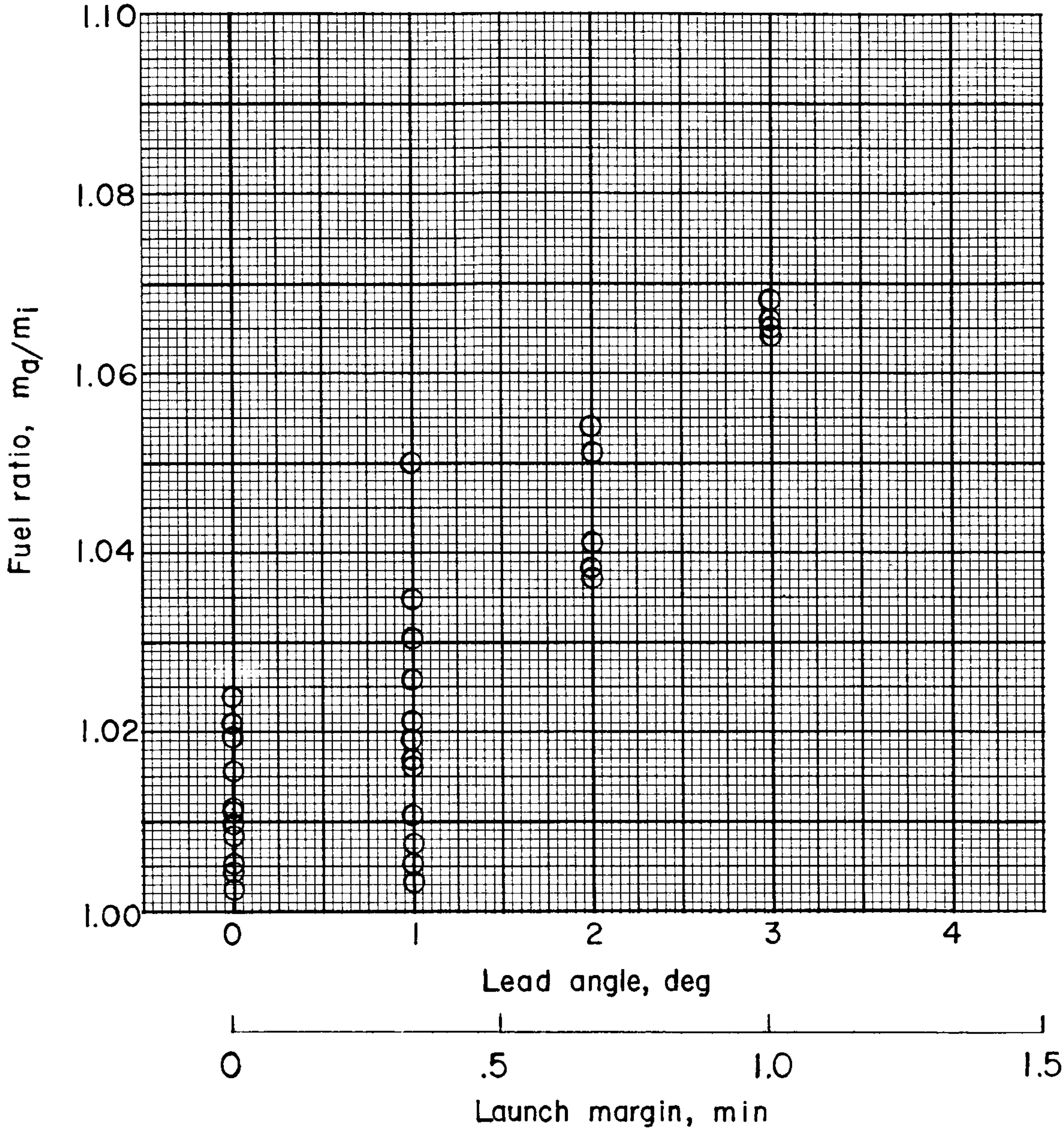
(a) 24° transfer.

Figure 14.- Results of on-time and early launches for pilot-controlled transfer trajectories.



(b) 90° transfer.

Figure 14.- Continued.



(c) 180° transfer.

Figure 14.- Concluded.

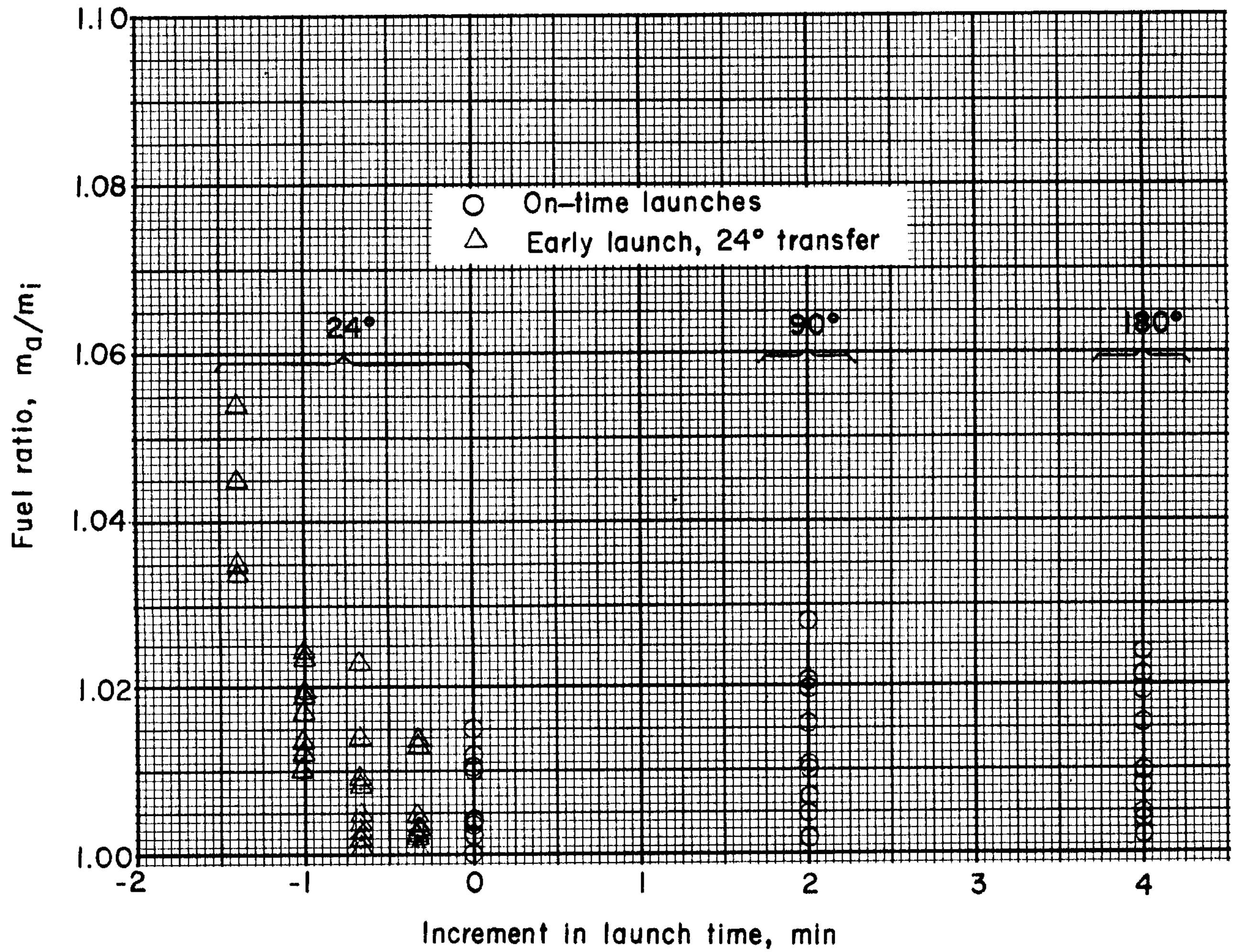


Figure 15.- Results of three transfer trajectories investigated relative to launch margin referred to 24° on-time nominal launch.